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RESEARCH AND DEVELOPMENT FOR PROJECT APOLLO SPACECRAFT (U)

NAS9-150

Volume III. Ground Qualification Tests

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INTRODUCTION

Volume III of SID 62-109 presents the Apollo ground qualification test plan which was briefly described in Section 3.0 of Volume I. The qualification test plans define the manner in which subcontractors and S&ID will accomplish that testing required to qualify subsystems for inclusion in the Apollo spacecraft and to qualify the associated ground support equipment. The integrated systems qualification test section defines the manner in which that testing required to qualify the total Apollo system for the successive missions will be accomplished.

The Apollo test program conforms with the intent of the statement of work, MIL-R-27542 and NCP-200-2 as modified by the negotiations with NASA during the past year. It reflects the philosophy delineated in the Apollo qualification-reliability test plan, SID 62-204, as modified by the reoriented qualification test program approved by NASA during the Big 4 exercise of November 1962 and as delineated in SID 62-1405.

The ground qualification test program has been planned for each subsystem to achieve qualification with minimum hardware testing through the delineation of subcontractor requirements and the integration with S&ID test programs into progressively advancing qualification status. This will permit complete qualification in simulated mission environments while eliminating duplication of testing at various assembly levels. This test plan will utilize design, development and qualification data from all test areas for equipment qualification and reliability assessment. Ground qualification of major subsystems, subassemblies or components peculiar to these subsystems will be done by the subcontractor. Responsibility for ground qualification for combined system compatibility and integration will remain with S&ID.

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1.0 SCOPE

1.1 PURPOSE

This document establishes the ground qualification test program for Project Apollo spacecraft and its associated GSE. This program's purpose is to prove the design intent and flight worthiness of each subsystem to and beyond specification limits of functional and environmental stresses by ground simulation of critical mission requirements. Programs, schedules, and environmental criteria are delineated for all Apollo spacecraft subsystems and associated ground support equipment. The individual qualification of each subsystem is meticulously planned by clearly defining the scope of each test program, delineating the tests to be performed with the hardware required, and establishing minimum test criteria consistent with equipment criticality. The ground qualification test program culminates in the operational integration of all subsystems as an entity to assure dimensional and functional compatibility.

1.2 OBJECTIVE

The objective of the ground qualification test plan is to present a program which delineates the philosophy to be employed to verify the capabilities of S&ID designed Apollo spacecraft subsystems and its associated GSE relevant to accomplishing the intended mission. Further, the implementation of this philosophy is presented in as much detail as is possible at this time. Ground qualification will be achieved by demonstration of system capability to perform within specification requirements and that functional performance and quality provisions are repeatable on randomly selected test articles. The program will show a properly oriented step by step qualification plan from one level of assembly to the next, and from one mission phase to the next. Ultimate qualification for the lunar mission is achieved just prior to the launch date. Qualification for each preliminary step in the flight program is achieved just prior to its relevant launch date.

1.3 CONFORMANCE

The ground qualification test plan is submitted as partial fulfillment of the Apollo documentation requirements delineated in NASA letter contract No. NAS9-150, dated 19 December 1961, entitled Research and Development for Project Apollo Spacecraft. The test plan conforms to the requirements of the Apollo spacecraft development statement of work, dated 18 December 1961, and has been guided by the provisions of NASA

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quality publication NCP 200-2, dated 15 December 1961, entitled Quality Assurance Provisions for Space System Contractors; and MIL-R-27542 (USAF), dated 21 May 1963. This volume details the reoriented Apollo qualification test plan as presented to and agreed upon by NASA MSC in November 1962.

1.4 REVISIONS AND AMENDMENTS

This volume will be revised and amended as required to reflect the current status of the qualification test plan of the project Apollo General Test Plan.

2.0 GROUND QUALIFICATION PROGRAM

2.1 QUALIFICATION PROGRAM LOGIC

The ground qualification program provides an optimum compromise from a statistically rigorous demonstration of all required design parameters to a reasonable program that can be done within development constraints. The program consists of a logical combination of tests designed to verify the elimination of all potential weak points or failure modes and to demonstrate the existence of suitable design margins to environment and time domains.

No attempt is made to demonstrate the reliability of the system or any of its components through conventional techniques. This is an impossible task within the time allotted for the program and reasonable financial limits. The program provides a good sound engineering approach toward insuring adequate system reliability. This is done by designing a development test program which will explore all of the potential causes of unreliability and eliminate them early in the program. The qualification tests verify that this process has been done, and that the components and systems will work well together in the anticipated environments.

The qualification test program is divided into two phases:

Phase A. Design proof tests
Phase B. Mission simulation life tests

The design proof phase is conducted in two parts. The first part follows procedures specified in MIL-STD-810 except that levels of environments applied are those commensurate with the anticipated maximum level for a typical Apollo mission. Each environment is applied at this maximum level, in sequence and in ascending order of severity to avoid prejudicing a test by what may be a deleterious effect of a prior test. These are applied at the component level to verify that it is able to perform the required function without failure under single worst case conditions.

Design proof tests are carried a step further in the second part of this phase where the off-limit test is applied to verify design margin. Both structural and functional design margins are verified here by a stress-to-failure test at the component level. Failure is defined as the inability to perform any required function. Critical failure modes are first determined through the failure mode analysis. The off-limit test insures an adequate design margin, thereby, reducing or eliminating all conditions that are potentially unreliable. This test when applied to all levels is actually the heart of the Apollo reliability program.

Phase B of the qualification test program consists of the mission simulation life test. It demonstrates system characteristics, but particularly, life under simulated mission conditions. This test includes the application of combined environments as facilities permit and provides the following essential data regarding the system performance as it affects reliability.

Component compatibility
Combined environment compatibility
Life characteristics
Repeatability

Through the application of the combined environments in the sequence and at the normalized levels anticipated for an actual mission, a comprehensive demonstration of the system capability is attained.

2.2 RELIABILITY ASSESSMENT LOGIC

The Apollo test program does not have as one of its requirements or for that matter an objective, the demonstration of the ultimate reliability of Apollo to any reasonable confidence level. Since the beginning of the program it has become obvious that this would be a task well outside the scope of time available and the milestones imposed on the program. It has been necessary to depart from the normally accepted practice of demonstrating reliability in a statistically rigorous manner. For Apollo, S&ID has adopted a new concept of reliability assurance where calculating and/or predicting failure rates of equipments from tests designed to accumulate operating time only becomes a secondary objective.

The Apollo qualification program does not yield the necessary data to show the over-all program reliability objectives to any assessable degree of confidence. A base line of reference must, therefore, be established in order to make an assessment of reliability at any given time. This base must apply to each separate functional area in order to estimate the potential reliability of a given function. For example, the group of components involved in providing the deep space communications (DSIF) may be considered a separate function and should be assessed independently of all other functions. Intelligent assessment below this level is impractical because of the low probability of a failure occurring. To describe this probability the expression mean- (or average) time-to-failure (MTTF) is the most applicable term for Apollo. If this must be related to the reliability objective associated with mission success, i.e., 0.96 it can then only be done by applying it

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at the functional level, such as: deep space communications, propulsion, dc power, etc. In most cases this is below the subsystem level. This will establish the base line of an 8400 hours mean-time-to-failure for the group of components making up a function. This applies where no redundancy exists exclusive of the grouping. With the base line established, it is now possible to state the ramifications relevant to the assessment of reliability within the qualification program.

The assessment of the reliability of the Apollo will be done without any extra test time or equipment. The reliability assessment program will be based on the use of attribute data from all phases of the testing activity incorporating data generated from the moment of design freeze to the moment before the final launch. Table 2-1 shows the data sources to be used in deriving an assessment, and the assumed percentage of data considered applicable from each source. When data becomes available from one of these areas, an assessment will be made (based on that data) and will be related to the 0.96 as applied to the function and to the spacecraft level. No confidence level will be given until late in the program when this may become possible.

Table 2-1. Reliability Assessment Data Sources

Source	Percent of Data Applicable for Reliability Assessment	Percent of Required Reliability Data Supplied
Development	10	5
Design proof	80	25
Mission life	100	40
Acceptance	40	5
Boilerplates	30	5
Spacecrafts	80	20

2.3 SUBSYSTEM QUALIFICATION PHASES - SPACECRAFT

2.3.1 Design Proof for Apollo

The Apollo design proof tests (DPT) are environmental tests conducted in sequence and at the maximum levels anticipated for a given environment

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anywhere and/or at anytime during a normal or abnormal mission if it is within the design criteria. Abnormal may be considered a situation such as max q abort where crew safety is essential and subsystem integrity a secondary consideration. The test safety margin used in these tests is about 1.33 times the level anticipated in more than 90 percent of the possible situations. The test safety margin is defined as a test level somewhat above the normally anticipated level for a given environment, but below the designer's margin. For example, if the normally anticipated vibration level for a given component was 1 g^2/cps and the component was designed to withstand only 1.5 g^2/cps the DPT level may be 1.33 g^2/cps.

The design proof tests are conducted in a MIL-STD-810 manner for all spacecraft equipment except that the levels imposed are those applicable to the Apollo mission. (See Section 4.0.) There are two exceptions which are applied to Apollo. The explosion proof test is applied to all components of the command module interior and is designed to simulate the oxygen atmosphere. Propellant vapors are imposed on the remaining components, but only when necessary.

The electromagnetic interference tests are designed according to S&ID specification MC 999-0002B.

One additional constraint imposed on the qualification test program is the special handling restriction applied to Apollo during the Big 4 exercise. This allowed spacecraft and associated equipment to receive any special handling needed to properly move, operate and preclude the detrimental effects of these actions and the associated weather. The test program takes advantage of this wherever possible, reducing the test requirements in those areas where the anticipated prelaunch environment was more stringent than the post launch counterpart. Typical of these environments are humidity, which is now only a requirement for some service module components, and shock, which is very light for equipment that need not work after reentry.

The program is based on the assumption that no failures will be encountered during the qualification tests apart from the off-limit tests. This is an optimistic point of view, but it does serve as a base line for the program. Any contingencies will change the nature and complexity of the qualification program relevant to the particular component involved. This approach concentrates the effort where it is needed, but also raises the requirement for a quick reaction capability when the contingency does arise.

The design proof tests will consist of, and be limited to the following applied environments, in sequence, at the levels delineated in Section 5 under specific subsystems:

2.3.1.1 Command Module and Associated Equipment

Vibration
Acceleration
Temperature extremes
Acoustics
Shock (two levels and types)
Vacuum
Electromagnetic interference
Explosion (O_2 atmosphere)

2.3.1.2 Service Module and Associated Equipment

All of 2.3.1.1 except oxygen
Humidity
Salt fog
Sand and dust

Minor exceptions to this criteria are applied to those items on the exterior such as umbilicals. The specific levels of environments applied are in Section 4.

2.3.2 <u>The Off-Limit Test</u>

2.3.2.1 Application

The off-limit test is considered a part of the design proof phase, but successful demonstration of a given design margin is not held as a contingency for qualification of any component under test. The tests will determine the design margin relevant to any critical failure mode. The intent is to verify that there are no cumulative tolerance situations which the tests indicate may contribute to unreliability.

2.3.2.2 Characteristics of the Off-Limit Test

The off-limit test determines design margins and further enables the engineer to predict the potential reliability of a given characteristic or component. There are two distinctly different types of Apollo off-limit tests and they are associated with verifying either stress or performance margins.

The first may involve stress to destruction. The tests are accomplished by determining the stress-strength margin related to each characteristic or failure mode and then algebraically summing them up. Some of this work will be done during the development program on the lower levels of assembly but there is a definite need to verify the results and to examine the effects of interfaces during qualification.

2.3.2.2.1 Off Limit Tests for Safety Margins. Figure 2-1 shows typical off-limit tests on a rocket motor. Two of the possible failure modes are analyzed by stressing the test sample to failure (see Part A, Figure 2.1). The case is pressurized to burst point, and the nozzles are operated until burn through occurs. If one of these conditions were depicted graphically and the test were conducted many times, the results would be much like Part B, Figure 2-1. The test should be conducted on ten or more samples for reasonable accuracy.

To satisfactorily analyze the failure mode, as the plot indicates, the characteristics of the stress, in this case pressure, though it could be any environment, and that of the strengths of all the components to be utilized must be known. In most cases the stress can be determined within fairly close tolerances. Knowing the accurate stress, the probability of a failure relevant to that stress is a function of the safety margin and the distribution of strengths associated with the stress. Since establishing an adequate design margin is not the complete answer, the verification of its magnitude and the assurance of a restricted distribution around the mean is also essential. In theory, it is possible to determine the safety or design margins, with a very few samples when distribution of strengths approaches a single value.

Since many samples at the subsystem level are impractical in Apollo, strict control measures must be used to limit the possibility of critical parameter variation to within very close tolerances. This is done during the design phase by careful design analysis at all assembly levels. Distribution control is further augmented by employing 100 percent screening of parts and material through the use of destructive and non-destructive testing techniques. For additional data see SID 62-109, Volume IV, Acceptance Test Plan.

If the parameter variations are held to close tolerances, and a reasonable safety margin exists between the expected stress and the known strength, then the predictable reliability of the unit is very high. In fact, it should be so high as to make it impractical to assess its integrity in terms of the usual mean time between failure because wearout would occur in most items long before encountering a mission degrading failure.

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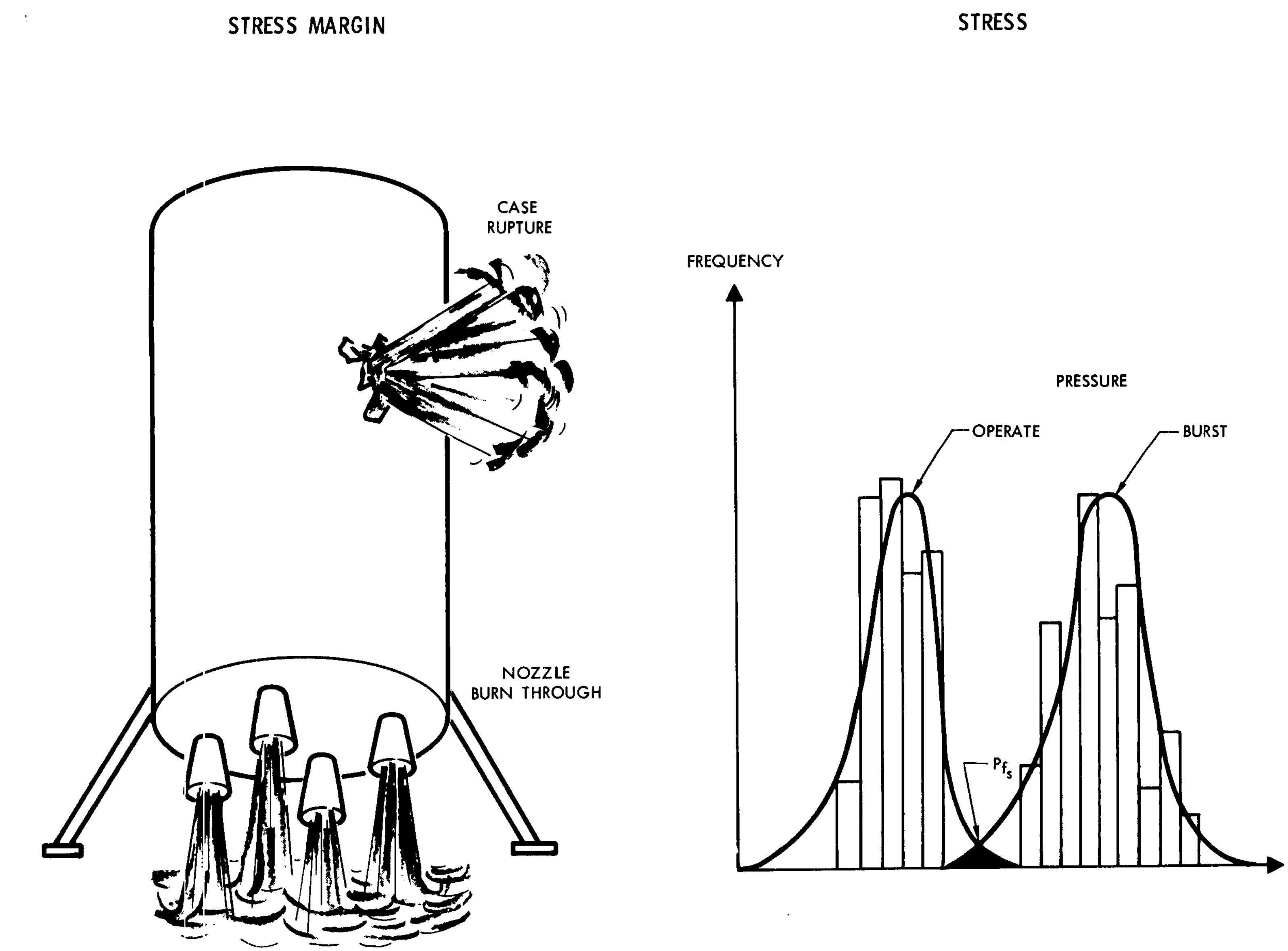

Figure 2-1. The Off-Limit Test for Safety Margin

2.3.2.2.2 Off-Limit Tests of a Function. The function off-limit test works in much the same manner as that used for determining margins of safety. (See Figure 2-2 where the rocket engine is being operated in a test stand.) This test determines the thrust impulse characteristic or operating limits of the engine. A plot of the characteristic is made after many firings. The performance margin is the difference between what is considered minimum acceptable and the mean of the distribution. This test is especially suitable for electrical and electronic equipment where performance variations must be expected, and be mutually tolerable within functions and subsystems.

2.3.2.2.3 The Off-Limit Test for Electronic Equipment. The off-limit test as applied to electronic equipment, is basically a functional off-limit test where design margins relative to line and input voltage variations are extremely important. If part parameters are held to close tolerances and a reasonable design margin is found during the off-limit test, component reliability will be very high, and impossible to predict by normal means. If malfunctions do occur they would probably be caused by a parameter strength distribution drifting toward the stress value (see Figure 2-3). Regardless of this, the actual probability of a failure is a function of the design margin which must be verified by tests.

2.3.2.3 Characteristics Examined by Off-Limit Testing

The exact characteristics to be tested for by off-limit stressing or test to destruction is a function of the failure-mode analysis which will be performed on every component used in the Apollo. The more critical modes will be analyzed by the off-limit techniques. Some characteristics to be explored during these tests are given in Table 2-2.

Table 2-2. Stresses Analyzed in Off-Limit Tests

Electronic Devices	Mechanical Devices
Temperature	Flow
Input Signal	Pressure Regulation
Supply Voltage and Power	Voltage Regulation
Vibration	Burst Pressure
Acceleration	Engine Thrust Chamber Burn-Through
Acoustics	Temperature

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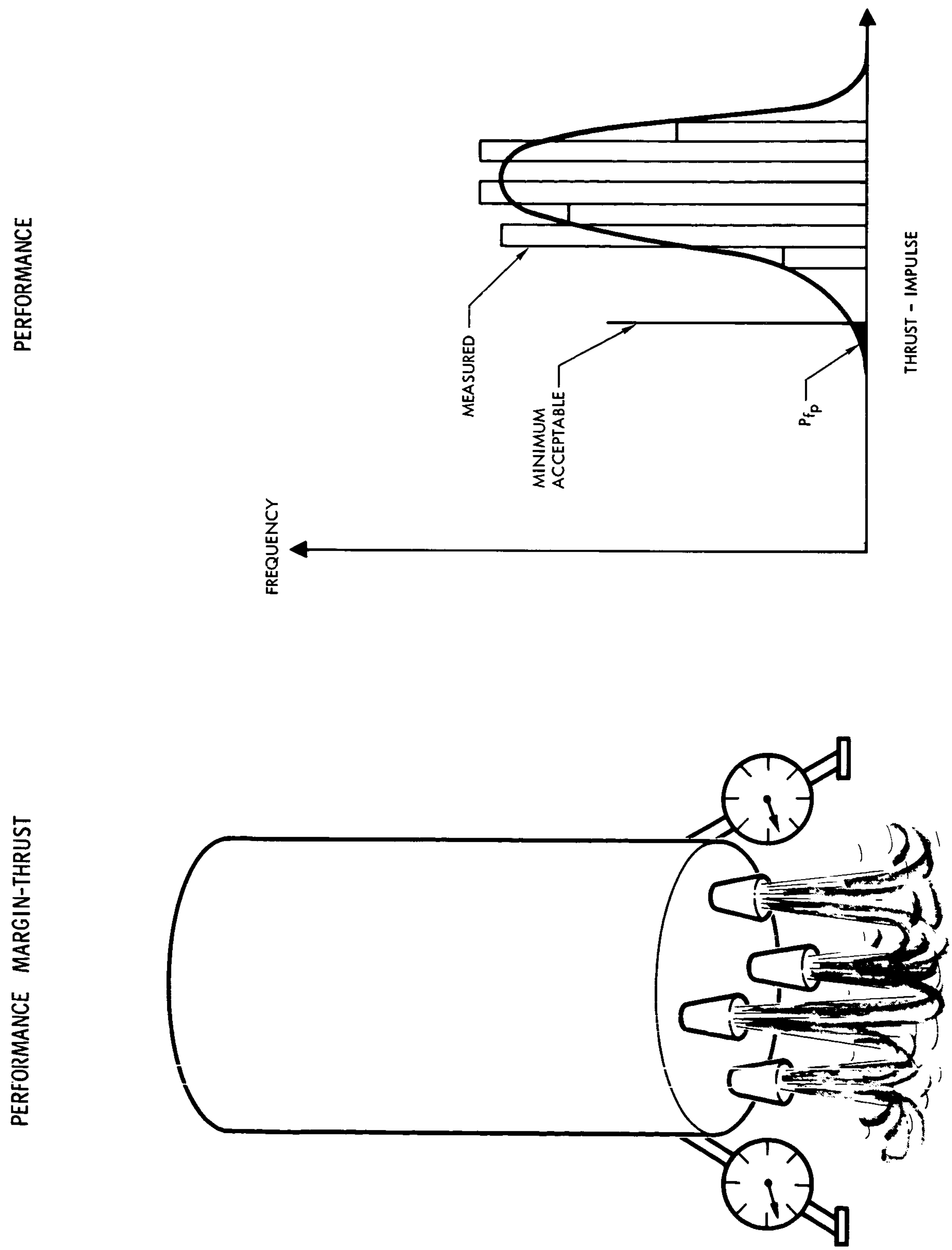

Figure 2-2. The Off-Limit Test for Performance Margin

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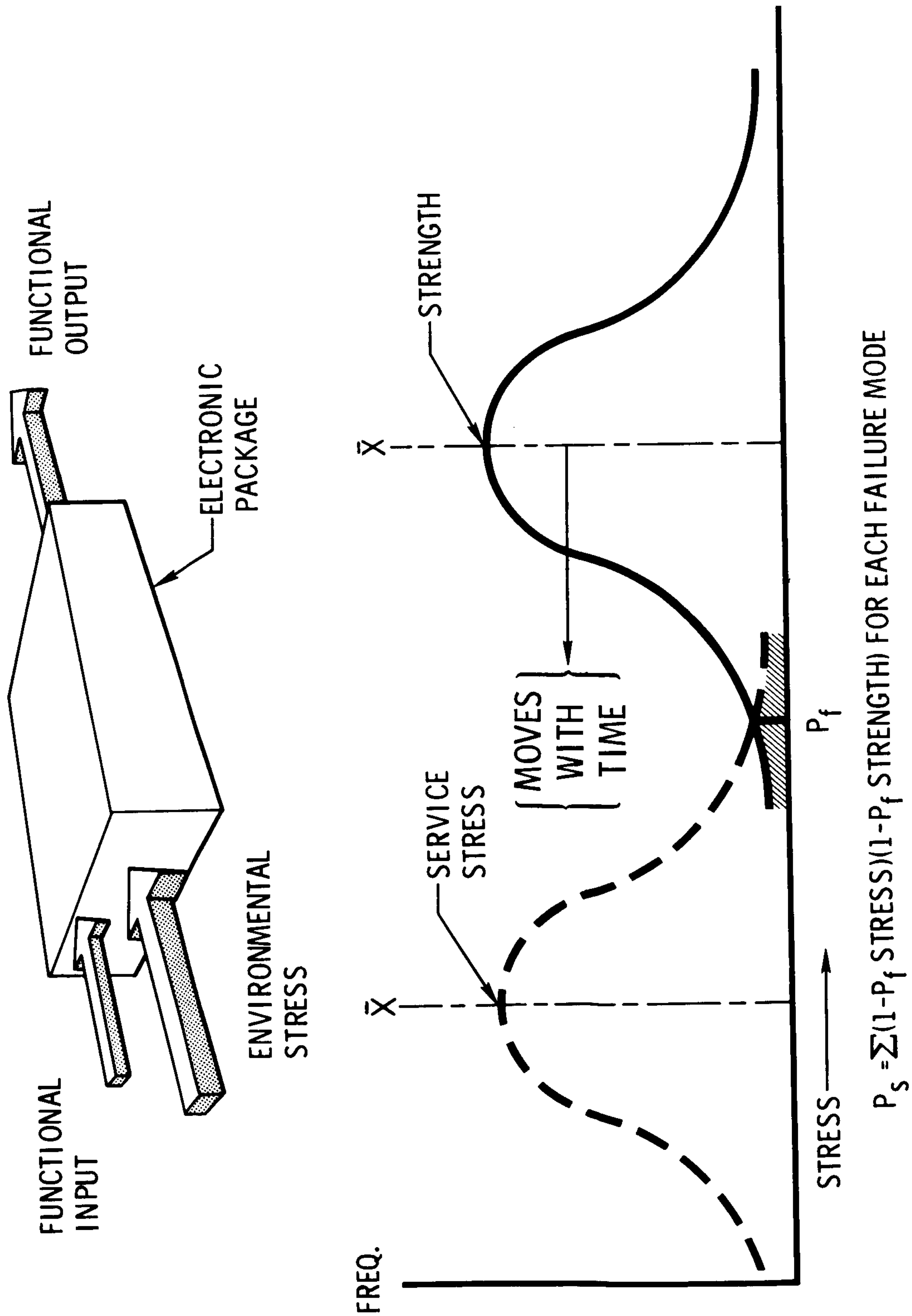

Figure 2-3. Reliability From Off-Limit Testing

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2.3.3 The Mission Simulation Life Test

2.3.3.1 Concept

The Apollo mission simulation life test simulates the normalized value of expected environments and the operational duty cycle for all phases of operation from the first checkout through recovery. The tests are to be conducted at subsystem or functional levels to insure that there are no component interface problems. They will include a natural combination of environments wherever practical, that is, where facilities and schedules permit.

Figures 2-4 and 2-5 depict the types of combined environments that may be expected relative to any given phase of the mission. Those pertinent to a mission simulation life test are:

Vacuum - Vibration
Acceleration - Vibration
Temperature - Vacuum

These may be produced in existing facilities, however, there are some weight and size limitations which may modify test planning.

The mission cycle (See Figure 2-6) is composed of two phases. One simulates activities during the prelaunch operations; that is from the time a unit completes fabrication until launch is initiated. The second phase simulates the post-launch period to and including the recovery operation. For the prelaunch phase all of the different testing and handling operations are simulated, including environments which are not compensated for by special handling, and the total aggregate test time anticipated for this phase. The post launch or in-flight time is simulated as closely as possible by the use of combined environmental facilities and the operation of the system in a manner which closely simulates the actual mission duty cycle.

For minimum-type systems, two complete mission simulations will be accomplished for qualification. Where more than one system is designated for qualification, up to two mission simulations will be utilized to satisfy this requirement. Where additional repeatability data are required, more mission simulation cycles will be performed.

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FOLD-OUT #1

	MATE C/M AND S/M	STATIC FIRING OF RCS	SPS	MOVE TO VACUUM CHAMBER	MOVE TO LAUNCH AREA	
DUAL SYSTEMS OUT	COMBINED SYSTEMS CHECKOUT AND PRE-FIRING CHECK 1.8 WK	FUEL CELL AND HIGH PRESSURE CHECKS 1.8 WK	0.2 WK	POST STATIC FIRING CHECKS 1.2 WK	VACUUM CHAMBER CHECKS 1.4 WK	INTEGRATED SYSTEMS CHECKOUT-COUNTDOWN AND LAUNCH 7.0 WK
OUT TIME S OF TEST /WK AY TIME 60 HR	CHECKOUT TIME 1 WK OF TEST 5 DAYS/WEEK 4 HR/DAY TOTAL TIME 20 HR	CHECKOUT TIME 2 DAYS 5 HR/DAY TOTAL TIME 10 HR	CHECKOUT TIME 1 DAY 4 HR/DAY TOTAL 4 HR	CHECKOUT TIME 1 WK 6 DAYS/WK 5 HR/DAY TOTAL TIME 30 HR	CHECKOUT TIME FUEL CELL 1 DAY 10 HR/DAY TOTAL TIME 10 HR ECS 2 DAYS 5 HR/DAY TOTAL TIME 10 HR SUIT CHECK 2 DAYS 5 HR/DAY TOTAL TIME 10 HR	CHECKOUT TIME 5.5 WK 6 DAYS/WK 5 HR/DAY TOTAL TIME 165 HR

Figure 2-4. Typical Apollo S/C Checkout Profile

FOLD-OUT #2

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	BOOST TO 100,000 FEET AND LET JETTISON	2ND STAGE BOOST 0.098	EARTH ORBIT STAY 4.500	TRANSLUNAR INJECTION 0.080	LEM TRANSFER 0.870	TRANSLUNAR COAST AND MIDCOURSE CORRECTIONS 76.130	LUNAR ORBIT INJECTION 0.090	LUNAR ORBIT COAST 41.830	EQUAL PERIOD ORBIT STAY, HOVER & TOUCHDOWN 6.290	LUNAR SUR-FACE STAY 32.000
ENVIRONMENTAL CONTROL SYSTEM	OPERATING ON FREON UP TO 100,000 FT	OPERATING ON WATER BOILER	OPERATION ON BOTH RADIATORS AND BOILERS EXCEPT DURING THE NIGHT AND/OR DARK SIDE AT WHICH TIME HEATING IS ACCOMPLISHED BY GLYCOL AND ELECTRONIC LOAD	OPERATING ON SPACE RADIATORS	OPERATION ON BOTH RADIATORS AND BOILERS EXCEPT DURING THE NIGHT AND/OR DARK SIDE AT WHICH TIME HEATING IS ACCOMPLISHED BY GLYCOL AND ELECTRONIC LOAD	OPERATION ON BOTH RADIATORS AND BOILERS EXCEPT DURING THE NIGHT AND/OR DARK SIDE AT WHICH TIME HEATING IS ACCOMPLISHED BY GLYCOL AND ELECTRONIC LOAD	OPERATION ON BOTH RADIATORS AND BOILERS EXCEPT DURING THE NIGHT AND/OR DARK SIDE AT WHICH TIME HEATING IS ACCOMPLISHED BY GLYCOL AND ELECTRONIC LOAD	OPERATION ON BOTH RADIATORS AND BOILERS EXCEPT DURING THE NIGHT AND/OR DARK SIDE AT WHICH TIME HEATING IS ACCOMPLISHED BY GLYCOL AND ELECTRONIC LOAD	OPERATION ON BOTH RADIATORS AND BOILERS EXCEPT DURING THE NIGHT AND/OR DARK SIDE AT WHICH TIME HEATING IS ACCOMPLISHED BY GLYCOL AND ELECTRONIC LOAD	OPERATION ON BOTH RADIATORS AND BOIL EXCEPT DURING THE N AND/OR DARK SIDE A WHICH TIME HEATING ACCOMPLISHED BY GL AND ELECTRONIC LOA
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Figure 2-5. Apollo LOR Mission Profile

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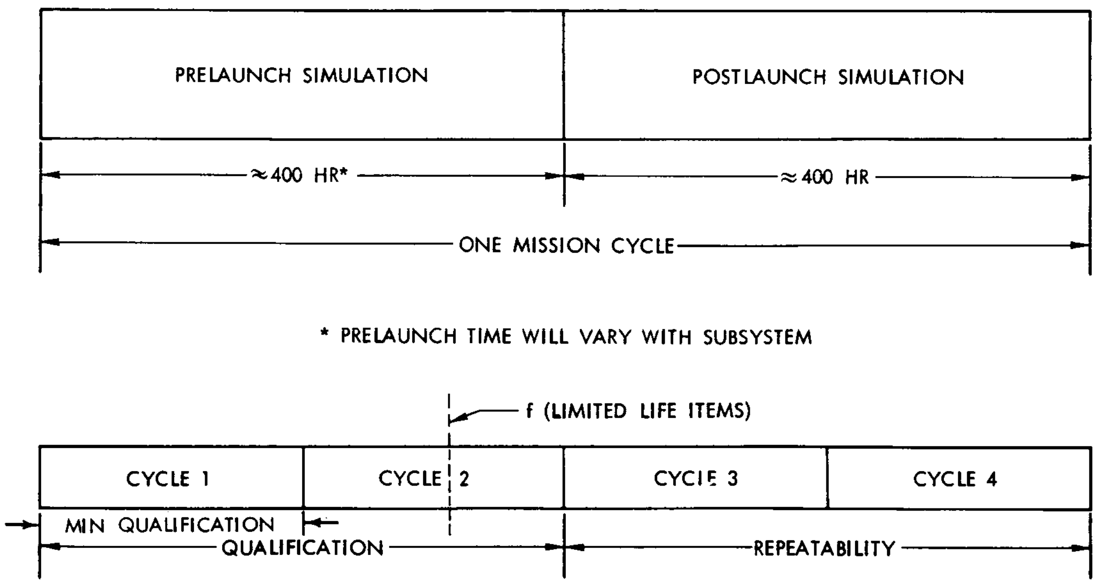

QUALIFICATION PROGRAM

Figure 2-6. Mission Simulation Life Test Profile

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The number of subsystems to be subjected to the mission simulation life test was determined a function of the criticality of the component function relevant to the intended mission and crew safety. The number of components required to support the systems in mission life tests were based on the expected life of the component. Therefore when any component within a given subsystem would reach its life limit before completing the test, a replacement was planned to assure four full test cycles.

2.3.3.2.1 The Requirement. The number of systems required to qualify a system under the mission simulation life test concept was determined to be a function of its criticality to the system and its anticipated duty cycle. Criticality is defined as a numerical index which relates the contribution of a given system to mission success and crew survival with all the other Apollo systems contributions. It is an approach toward equalizing the risk in testing and insures test emphasis in the more critical areas, but in proper relation to its contribution. The concept of criticality is covered in paragraph 2.4 of this volume. Its application to a specific subsystem is covered in the paragraph on the relevant subsystem in Section 5. In order to understand the application at this point, it is sufficient to state that the higher the criticality indices and the higher the duty cycle, the more tests are required to qualify the system. Much of the data required to complete the mission-life test analysis will be derived from attribute data obtained from other phases of the Apollo test program. It was necessary to use this data because of the magnitude of data required to qualify Apollo for the lunar flight.

2.3.3.2.2 Accumulated Operating Hours. The operating hours accumulated on a system from its fabrication to recovery were synthesized by considering every phase of component life. A mission profile was prepared depicting the prior-to-launch phases (see Figure 2-4) and the post launch phases (see Figure 2-5). These figures were used to derive the individual subsystems or function operating time for prelaunch and duty cycle for post-launch. The total operating time to be imposed during each mission life cycle on a given subsystem is the sum of both. The total for any given subsystem will be different and is given in the relevant paragraph of Section 5. Some typical examples are shown on the profiles. The environments expected in each phase and to be applied during the tests are covered in detail in Section 4.0.

2.3.3.2.3 Attribute Hours. The attribute hours used to reduce the over-all tests required to assure the mission life capability are derived in much the same manner as those used for the assessment of reliability. Table 2-1 may be used to determine the percentage of useful data obtained from these sources. This alone does not provide the necessary data to accomplish qualification to the mission life requirements, because there is still doubt concerning the validity and applicability of the data sources. It

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was also necessary to establish a minimum program for the mission life tests. In the minimum program, strict control could be exercised and a base of reference established to compare with other data to assure validity. The minimum mission life test is given in paragraph 2. 5.

2.4 THE CONCEPT OF CRITICALITY

2.4.1 Logic

The logic used to derive numerical values for criticality, and the functional relationship between criticality and test time, is predicated on the following assumptions:

a. Product integrity, as measured by the mean-time-before-failure (MTBF), may be extended as the amount of experience acquired from testing is increased. MTBF may, therefore, be considered a function of the test time. This presupposes that testing reveals information which leads to design improvements, process or quality changes, etc. , all of which result in improved design integrity.

b. More testing is required on systems of higher criticality than on those of lesser criticality.

c. More testing is required on systems which cannot be improved through analytical techniques.

d. More testing is required to verify the design integrity of components that represent recent advances in engineering sciences.

Criticality is a function of the relative orders of redundancy, alternate modes of operation, spares incorporated in a given subsystem design, and their effect on crew survival or mission success. Therefore, test time on subsystems composed mainly of series components must be greater than that required for subsystems with parallel components or where spares and/or alternate modes of operation are provided.

The intent is to reduce the risk that high criticality or series components will cause or contribute to loss of the crew or failure to accomplish the mission. The program provides a test where there is no greater risk factor with critical items than there is with hardware of lesser criticality. Equalizing risk or establishing a constant risk factor where an optimum approach is the objective (as is the test program outlined herein).

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2.4.2 Derivation

2.4.2.1 The Indices

The criticality indices were derived on the basis of equalizing risk. Risk can be defined as crew or system losses per mission. Mathematically it can be expressed as:

$$\text{Risk} = \frac{\text{crew or system losses}}{\text{component failure}} \times \frac{\text{component failures}}{\text{mission}} \qquad (1)$$

The first term of this expression, crew or system losses/component failure, defines the criticality of the component with respect to its influence on crew survival or success of the spacecraft. The second term, component failures/mission, is inversely proportional to the MTBF. Introducing these relationships into 1 results in the following equation:

$$\text{Risk} = \text{Criticality} \times \frac{C}{\text{MTBF}} \qquad (2)$$

where

$$C = \text{A proportionality constant}$$

From the functional relationship between MTBF and test time

$$\text{Risk} = \text{Criticality} \times \frac{C}{f(\text{test time})} \qquad (3)$$

For a constant risk, C divided by the constant risk results in a new constant which can be symbolized by C'. Equation 3 then reduces to:

$$\text{Test time} = C' \times f\,(\text{criticality}) \qquad (4)$$

Before equation 4 can be translated into practical terms, the criticality equation must be derived. Repeating the previous definition:

$$\text{Criticality} = \frac{\text{crew or system losses}}{\text{component failure}} \qquad (5)$$

Equation 5 can be written in terms of partial derivatives of probabilities of failure, as follows:

$$\text{Criticality} = \frac{\partial P_{\text{crew or system loss}}}{\partial P_{\text{component failure}}}$$

$$= \frac{\partial P_{\text{crew or system loss}}}{\partial P_{\text{module failure}}} \times \frac{\partial P_{\text{module failure}}}{\partial P_{\text{component failure}}} \qquad (6)$$

But, failure = 1-success, and $\partial P_{\text{failure}} = -\partial P_{\text{success}}$; wherein the probability of success is equal to reliability R, and equation 6 can be restated in the form:

$$\text{Criticality} = \frac{\partial P_{\text{crew survival or mission success}}}{\partial R_{\text{module}}} \times \frac{\partial R_{\text{module}}}{\partial R_{\text{component}}} \qquad (7)$$

The latter permits a numerical evaluation of the effect of component reliability on mission success or crew survival. Criticality is, therefore, the rate of change of spacecraft reliability or crew safety with respect to the reliability of any component.

2.4.2.2 Sample System

As an example of the application of equation 7, consider the helium supply within the service module reaction control system. A schematic of one of four modules within the spacecraft is shown in Figure 2-7. From the referenced schematic, the reliability logic diagram of Figure 2-8 may be synthesized. With the four module quad concept employed in the design of the reaction control system, mission success is assured if any three of the four modules function satisfactorily. Crew survival is assured if any two of the four modules are functioning. These criteria can be translated into an equation for probability of crew survival, which is equal to the sum of the probabilities of four, three and two modules operating, or for R_M = module reliability:

$$P_{\text{crew survival}} = R_m{}^4 + 4R_m{}^3 (1-R_m) + 6R_m{}^2 (1-R_m)^2 . \qquad (8)$$

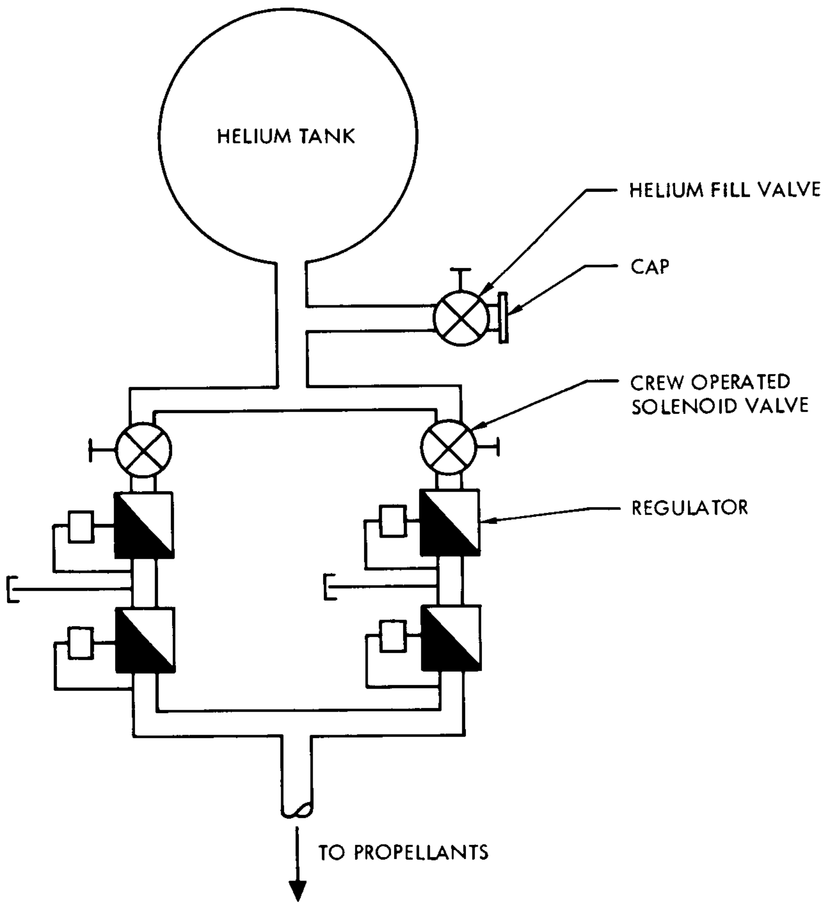
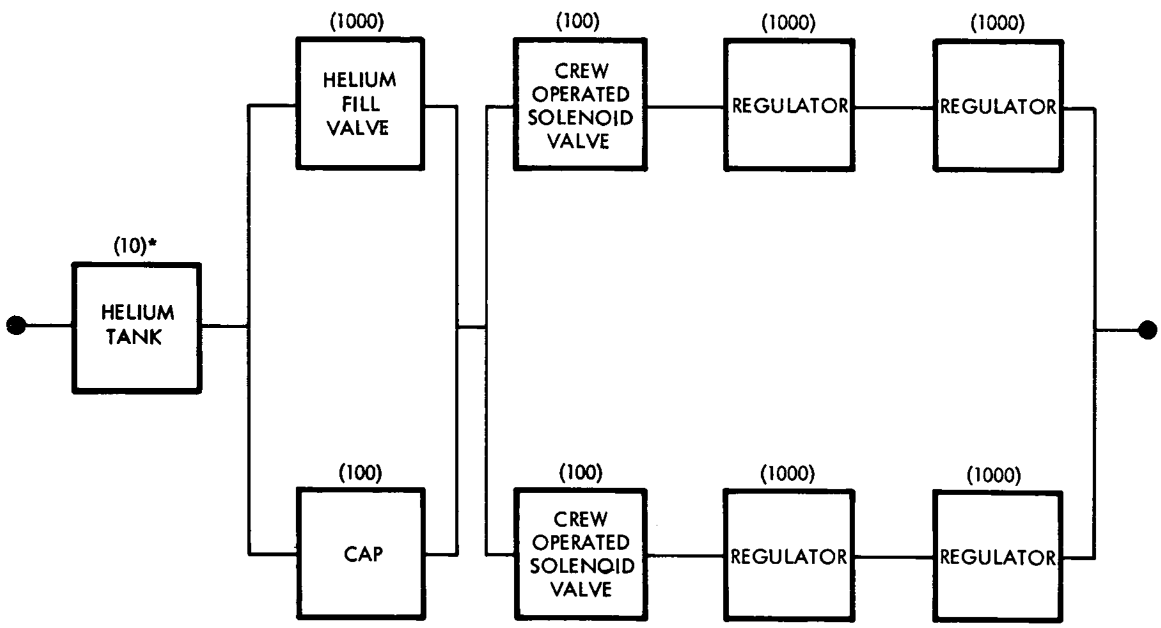


Figure 2-7. Service Module Reaction Control System Helium Supply Schematic (One of Four Modules)



*PARENTHETIC NUMBERS ARE FAILURES PER 10^6 MISSIONS

Figure 2-8. Reliability Logic Diagram Service Module Reaction Control System Helium Supply

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The partial derivative of the probability of crew survival with respect to the reliability of the module is given by

$$\frac{\partial P_{\text{crew survival}}}{\partial R_m} = 12R_m^3 - 24R_m^2 + 12R_m \tag{9}$$

Equation 9 represents the criticality of the module with respect to the spacecraft. When the reliability of the module ($R_m = 0.987$) is substituted in equation 9, the following is computed:

$$\frac{\partial P_{\text{crew survival}}}{\partial R_m} = 0.00207 \tag{10}$$

Similarly, when the criticality of the crew operated helium valve with respect to the module is calculated, employing a valve reliability of 0.999989:

$$\frac{\partial R_m}{\partial R_{\text{He solenoid valve}}} = 0.00193 \tag{11}$$

The criticality of the crew-operated solenoid valve with respect to the spacecraft is, therefore, the product of the results of 10 and 11, or:

$$\frac{\partial P_{\text{crew survival}}}{\partial R_{\text{He solenoid valve}}} = \begin{matrix}\text{crew operated solenoid valve} \\ \text{criticality}\end{matrix} = 0.000004 \tag{12}$$

2.4.3 Application

2.4.3.1 Classification

Upon calculating the criticalities of the major components of the spacecraft, it was obvious that the numerical values could be grouped into three basic order-of-magnitude classes, with a gross simplification in determining the functional relationship between test time and criticality, as expressed in equation 4. . Additional justification for grouping criticalities was found in the fact that items with similar order-of-magnitude values not only contained similar degrees of redundancy, alternate modes, etc., but also were responsible for similar ramifications in crew survival or mission success. The criticality was grouped into the classes as shown in Table 2-3 with definitions and classifications.

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Table 2-3. Criticality Classification

Class	Index Range	Possible Effect on Mission	Functional Relation
1	0.8 to 1.0	Impair crew survival (catastrophic)	No redundancy
2	0.01 to 0.08	Impair mission success or abort	Dual redundancy
3	Less than 0.008	Maintenance problem or nuisance	Triple redundancy of some form or non-essential equipment

2.4.3.2 Application to Test Time

The transition from the criticality classification to the choice of an optimum test time must be based on sound engineering practice and program constraints. With this and the bare minimum program of paragraph 2.5 as the objectives, the plan depicted in Figure 2-9 was devised.

The plan presented closely conforms to the AGREE [1], task group 3 definitions for absolute minimum test time. The AGREE report recommends that the minimum test time be equal to or greater than three times the required MTBF in order to:

1. Provide additional data in case of dispute

2. Provide information for corrective action in case equipment is rejected

This level is also subscribed to by industry as a whole and is made a requirement in most present military specifications. It is based on the premise that if the reliability or mean-time-before-failure (MTBF) is much greater than the time during which it is tested, the chances of a failure occurring during the prescribed test cycle are negligibly small. All of the subsystems associated with Apollo fall into this category since the required reliability is between one and three orders of magnitude greater than the proposed test level.

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1 Reliability of Military Electronic Equipment. The Advisory Group on Reliability Electronic Equipment Office of the Assistant Secretary of Defense (Research and Engineering) June 1957.

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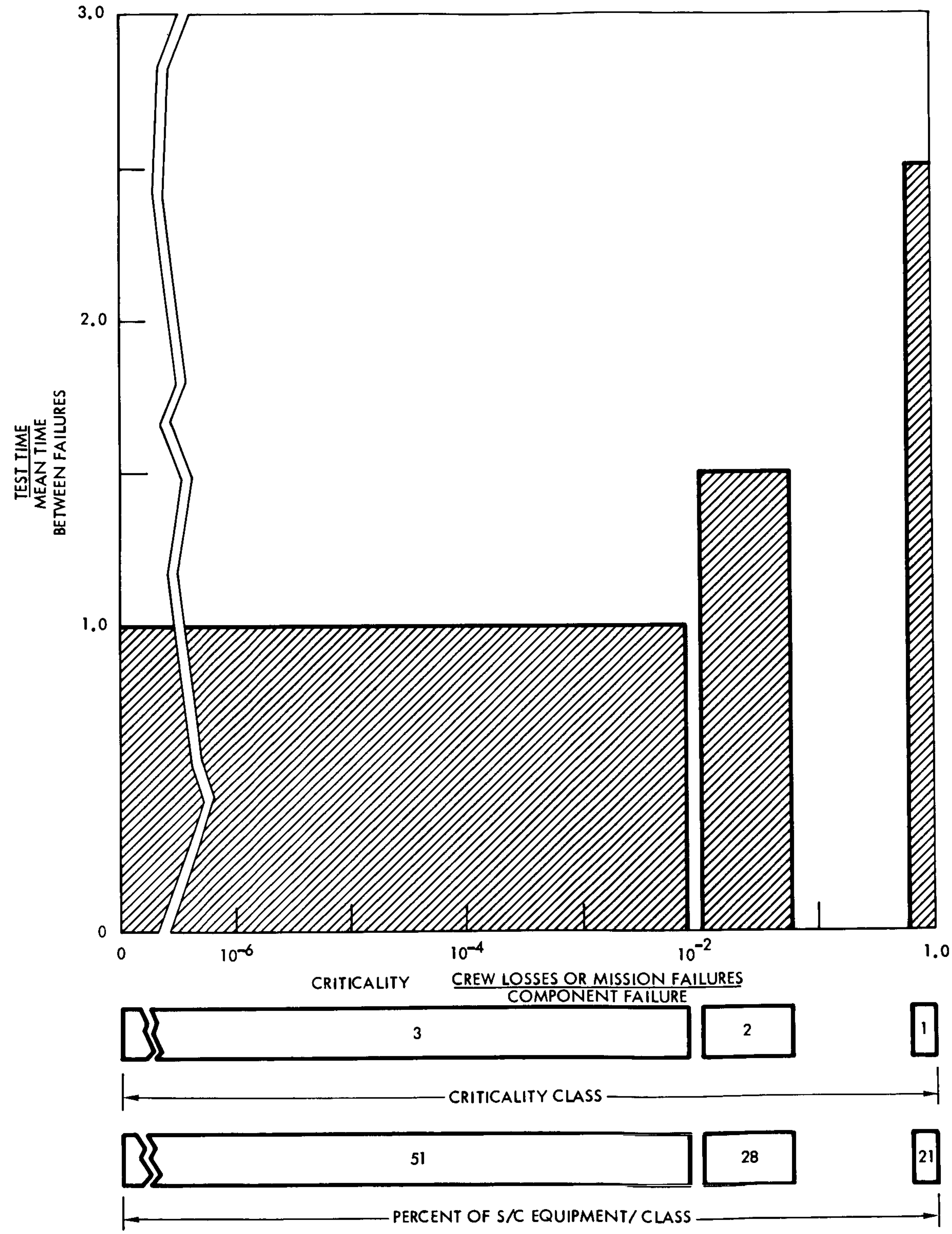

Figure 2-9. Criticality Versus Total Test Time

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The plan calls for imposing a somewhat lower than minimum program proposed by AGREE on the most critical items, those in criticality class one. Specifically, 2.5 times the mean-time-before-failure (MTBF) was chosen because it can be shown that it is the lowest possible test time that can be imposed on a system and have any reasonable assurance that it will approach the objective. This is true only if no failures are encountered anytime during the formalized test period.

For class 3 items a much more conservative program has been established. It was arbitrarily established at one MTBF on the premise that if it could not make this it certainly could not comply inherently with the program objectives for the component.

For criticality class 2 components, the program is established based on an objective of 1.5 times the MTBF, representing a reasonable compromise between the two extremes.

2.5 THE MINIMUM QUALIFICATION PROGRAM

In addition to the criticality constraint a minimum qualification program was established to represent a base point on which all subsystem qualification programs could be built. Anything less than this would compromise the objectives of the program and yield little intelligible data relevant to the qualification of the subsystems. The minimum program utilizes multiple hardware usage. It also assumes that no failures will be encountered. All failure modes that are potentially critical must be designed out of the systems and the tests must verify that the problem does not exist. The tests are planned and scheduled carefully to insure that a previous test conducted on a component will not prejudice the results of a following test. Refurbishment of components and systems is planned where multiple tests are scheduled and detrimental effects expected during the early phases.

The minimum qualification program was defined to include two components or lower levels of assembly, for sequential single environmental, explosion proof and electromagnetic interference tests. One component or lower level of assembly was required for off-limit tests. One additional subsystem was scheduled for mission simulation and life tests, with necessary replacements on high criticality, limited life and other items which required refurbishment prior to completion of any test phase. Table 2-4. summarizes the minimum hardware requirements.

Before they were incorporated into the qualification program, hardware requirements, above this minimum program, were justified on the basis of criticality, expected mission-life and service life. Since service life may cause the minimum program hardware requirements to vary considerably, service life has been selected and defined very closely.

[redacted]

Table 2-4. Minimum Hardware Per System

Test Phase	No. of Black Boxes and Lower Assemblies	No. of Subsystems
Phase A. Design proof		
Part 1. Environmental	2	0
Part 2. Off-limits	1	0
Phase B. Mission simulation and life	0	1*
Total	3	1*
*Plus replacements on critical and limited life items Refurbish others requiring such action		

2.6 QUALIFICATION PROGRAM CONTINGENCIES

The Apollo qualification program is considered a bare minimum and, therefore, will be carefully controlled by Apollo reliability to assure maximum applicability of each test and the data derived from it.

The probability and effects of a failure have not been provided for in the selection of numbers of hardware for tests and facilities planning. Any failure during qualification is considered a contingency which will necessitate additional hardware, test time and facilities rescheduling. It is most probable that failures will be encountered but to provide accent on the needy areas, planning for them must wait until failures are encountered. It is anticipated that as a failure occurs, a preliminary analysis will be made immediately. The program will be designed to determine the cause and insure proper corrective action. In order to reach this objective, a fast reaction cycle must be set-up between NASA and S&ID for the resolution of contingencies not included in the program, and to prevent serious slippages in schedules.

Reliability demonstration to any level of reliability or confidence is not considered a prerequisite to qualification and is beyond the scope of this program. An assessment will be attempted when time and data permit. It is anticipated that there never will be enough data to accurately determine the Apollo system reliability before flight time.

The test program, as applied to ground qualification of flight articles, is planned so that as each progressive step in the flight test program is executed, the individual components will have undergone some testing that will facilitate a decision as to its flight worthiness. Individual launch constraints corresponding with each intended flight article have been generated to define what must be accomplished relative to qualification of each component scheduled for the boilerplate or spacecraft prior to launch.

2.7 TESTING CRITERIA

2.7.1 Test Equipment

Equipment measuring item parameters will not introduce an error greater than ten percent of the tolerance on the parameter. If a parameter tolerance is plus or minus 10 percent, the equipment error will not be greater than plus or minus 1 percent of the parameter nominal value.

2.7.2 Failure Reporting, Analysis and Feedback System

Detailed malfunction data will be recorded as a part of normal equipment testing operations. All failures will be analyzed for corrective action requirements. The designation, retest OK will not be considered acceptable. A complete analysis will be performed on all failures to determine the cause of the failure or the reason for the symptoms reported.

2.7.3 Adjustment and Repairs During Test

No adjustments, repairs or maintenance will be done during tests unless they are part of a scheduled maintenance plan or not due to faults in design, materials, workmanship, or test conditions imposed. This condition will be imposed on all equipment during qualification. After qualification is accomplished this will be modified for mission simulation life tests.

2.7.4 Waiver of Tests

If evidence can be furnished of equipment or component compliance to requirements of the procurement specification, or evidence is provided certifying that similar equipment or components have successfully passed tests to equivalent or greater criteria than those required, the applicable portion of these tests will be waived. The required evidence to support partial or complete qualification by similarity includes the following:

1. Complete description of similar item, including photographs, drawings and performance data

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2. Test reports and test data of previously conducted functional and environmental tests (actual and similar equipment)

3. Supplier's comparison of the proposed article and that for which similarity is claimed; listing reasons for similarity claims

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3.0 QUALIFICATION SCHEDULES*

3.1 CONCEPT

The qualification schedules are graphical presentations of the anticipated test status for components of all Apollo subsystems. They show the time phasing of the Apollo master development schedule (see Figure 1, Volume I), with respect to boilerplate and airframe vehicle launch dates, and the phasing of component qualification relative to system qualification. The intent is to present test schedules in conjunction with flight schedules in order to foresee the need for launch constraints for each flight article. These constraints should specify the degree of success in development and qualification tests that each component must meet to assure reasonable success for the boilerplate or airframe mission.

3.2 APPLICATION

3.2.1 Explanation

The following describes how the qualification schedules foresee the need of launch constraints for a particular mission.

Prior to the mission, to the extent that ground qualification permits, it is desired to ascertain if a certain component has been qualified, in order to proceed to the flight qualification level.

The component in question is located in the applicable system schedule. The boilerplate or airframe launch dates, indicated by the labeled vertical lines, are compared with the phase of component qualification. If the launch date falls within the component development testing phase, constraints, consisting of a statement setting a minimum standard to be reached such as a success to failure ratio, will establish the degree of qualification required for that component. A launch date falling within the qualification test phase would require similar constraints to be established for components in question. If a component has completed qualification prior to the launch date, launch constraints will not be required.

3.2.2 Example

B-12 scheduled for flight in October 1963, will serve as an example. It will contain components of the launch escape, earth landing and structural

*Entire section reissued.

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subsystems. In the LES the launch escape motor, (see Figure 3-4) has completed 17 development tests, but no qualification tests. Similar conditions apply to the jettison and pitch control motors with 23 and 28 development tests, respectively. Including the amount of testing accomplished prior to launch, S&ID will state the degree of success to be met by each component, commensurate with assurance of flight success. The abort sequencer, tower structure, tower insulation, and explosive bolts will be treated similarly. The components of the Earth Landing System (see Figure 3-5) and Structures Subsystems (see Figure 3-13), shown as onboard equipment for B-#23, are qualified prior to the launch date and therefore do not require any launch constraints.

3.3 SUBSYSTEM SCHEDULES

(Note: Schedules will be expanded as additional information becomes available).

3.3.1 Service Propulsion System

Figure 3-1 shows that the SPS will be on-board AFRM 009 and AFRM 011. The SPS will have completed qualification prior to installation.

3.3.2 Reaction Control System

From Figure 3-2, it may be seen that the C/M RCS will be installed on B-22 and AFRM's 009 and 011. The system will be qualified for use prior to the launch date of B-22. Those components without schedules have not been released for manufacture.

Figure 3-3 shows that the S/M RCS is used on AFRM 009 and 011 only and will be qualified for those flights.

3.3.3 Launch Escape System

Examination of Figure 3-4 reveals the fact that launch constraints will be required for boilerplate flights B-6, B-12, B-13, B-23, and B-15. Although the jettison motor and hot wire initiators are qualified for use on B-15, the system is not qualified for use until the flight of B-22.

3.3.4 Earth•Landing System

Figure 3-5 illustrates the fact that the ELS is installed as a system on B-6, B-12, B-23, B-22, and AFRM's 010, 009, 002, and 011. The system will be qualified for use prior to the launch of B-22.

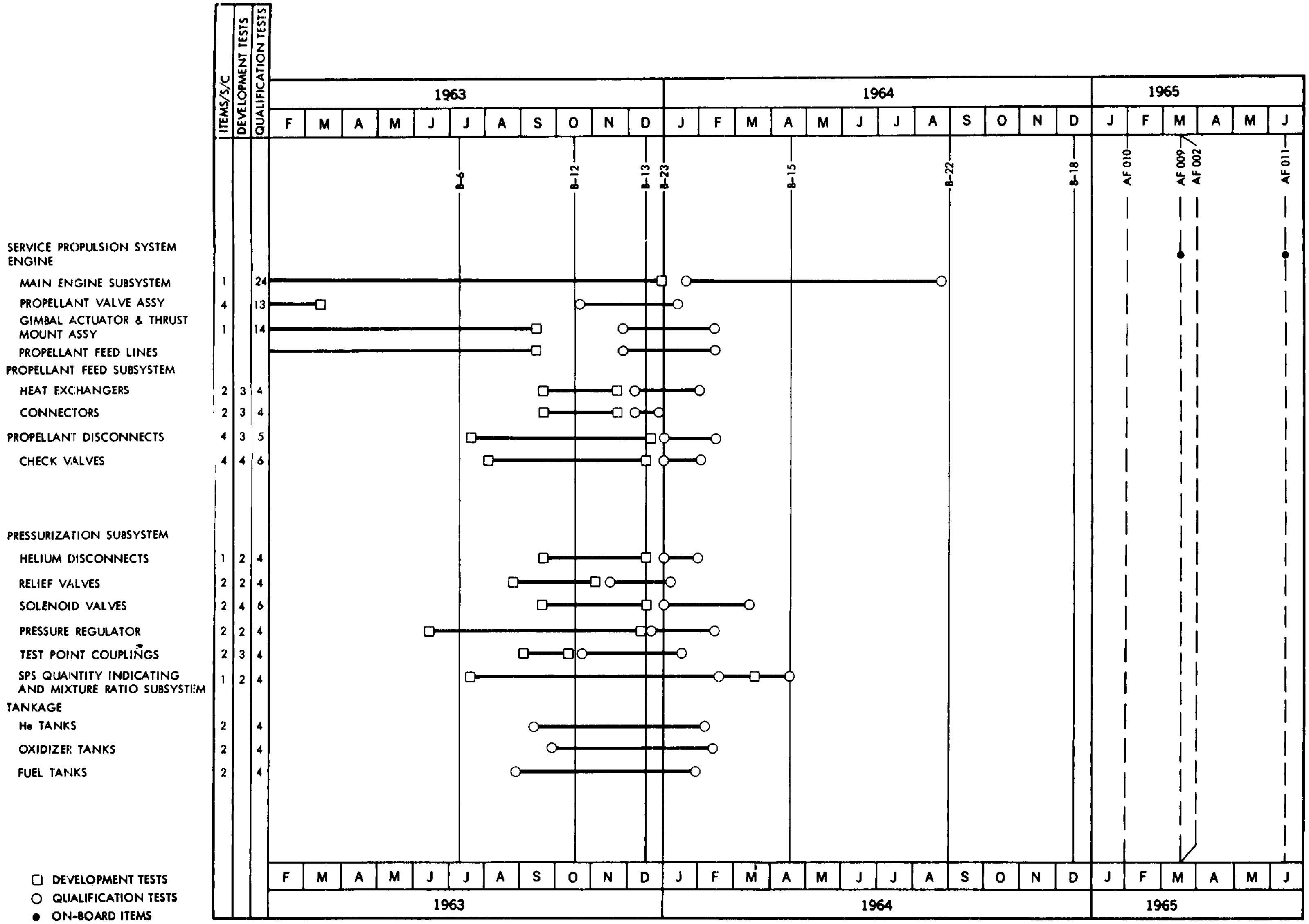

Figure 3-1. Service Propulsion System Component Schedules

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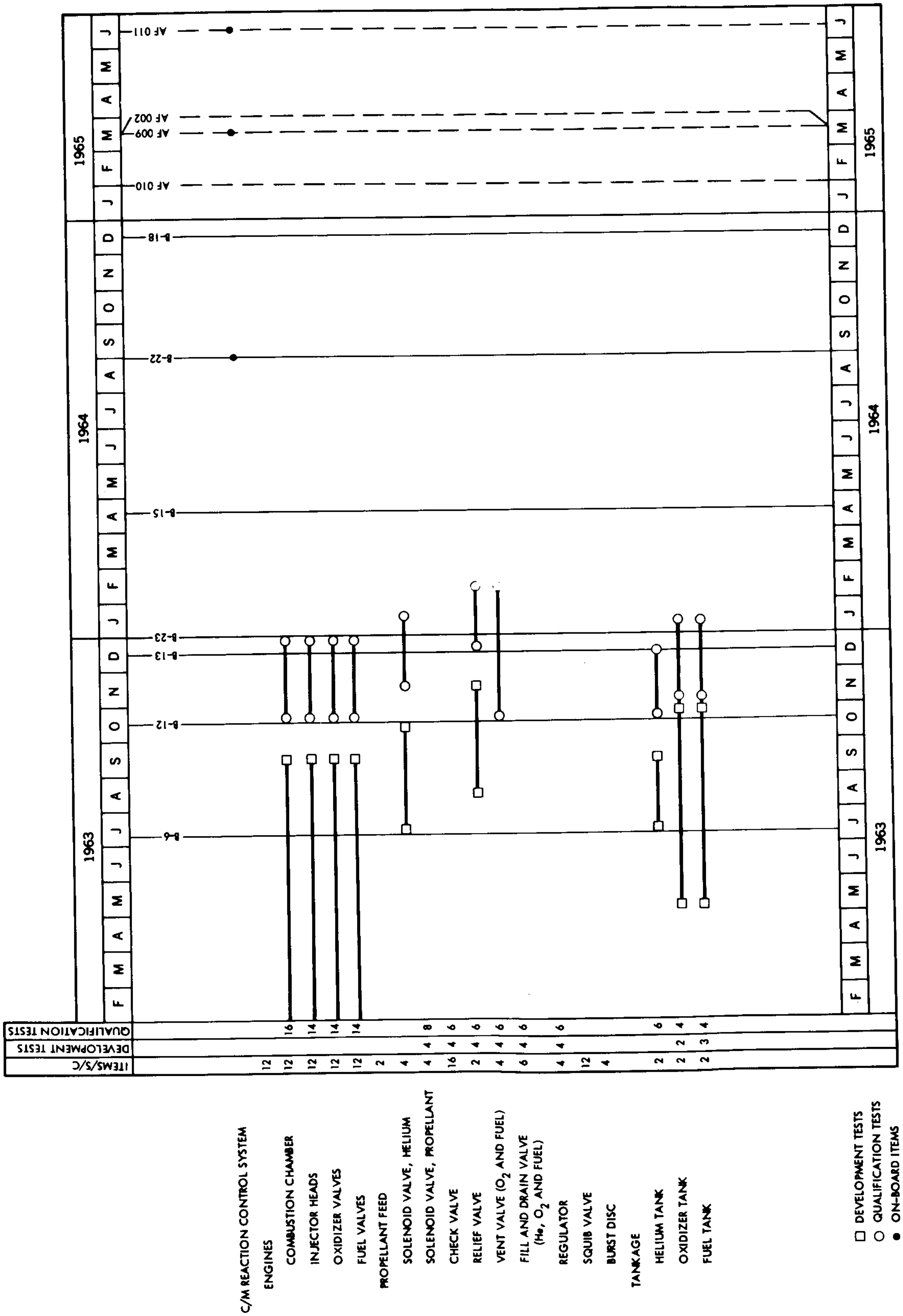

Figure 3-2. C/M RCS Component Schedules

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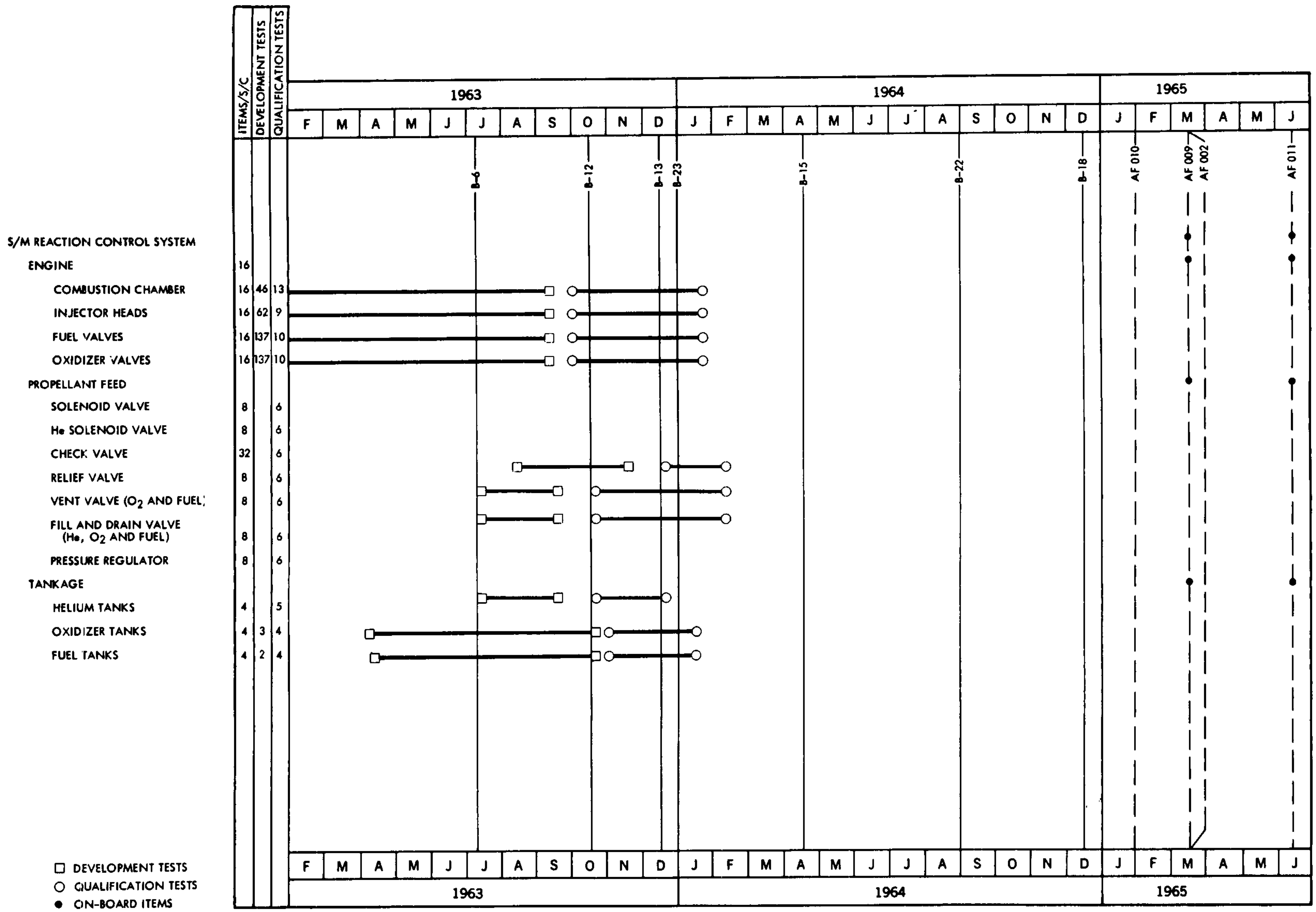

Figure 3-3. S/M RCS Component Schedules

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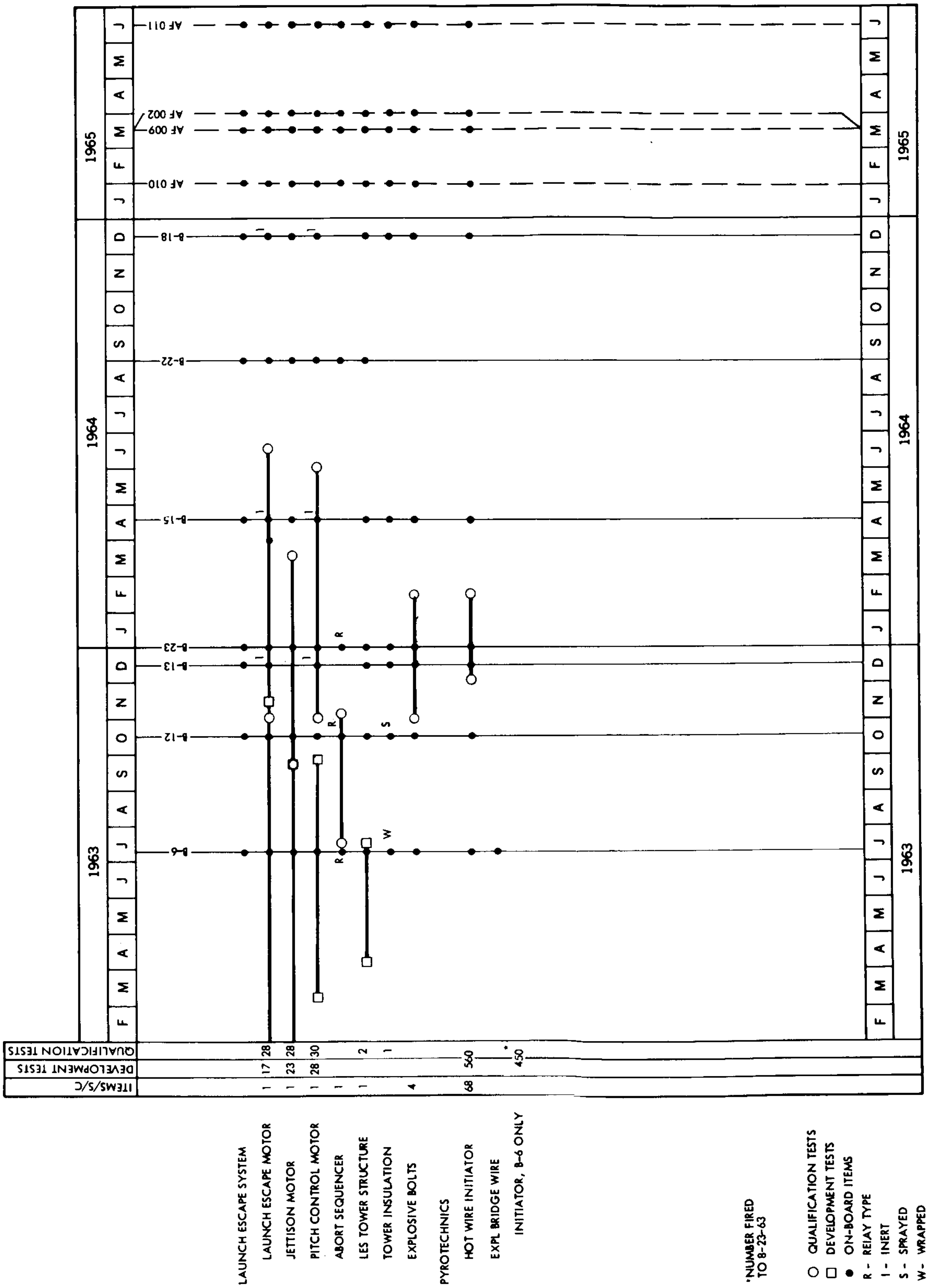

Figure 3-4. LES Component Schedules

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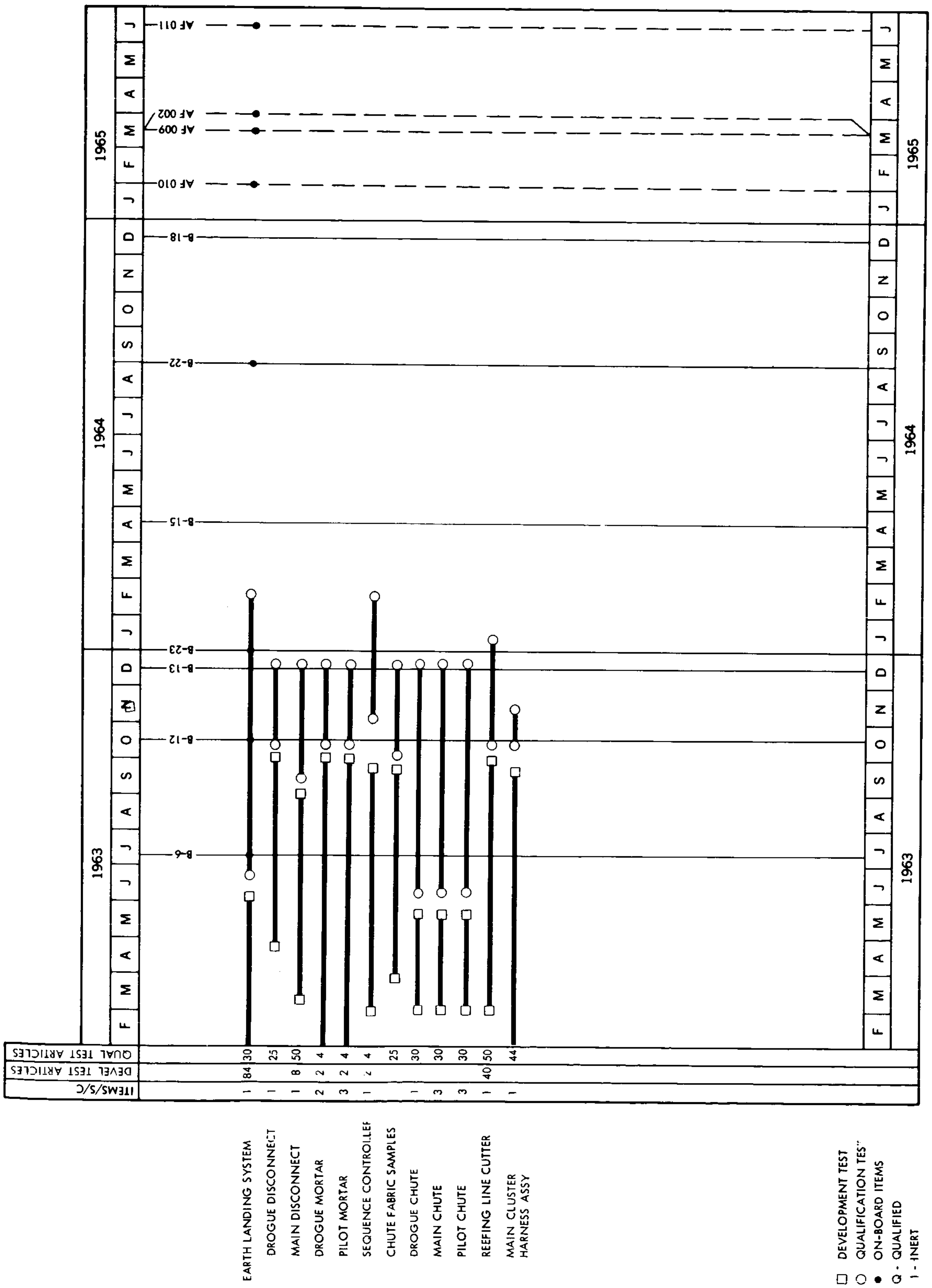

Figure 3-5. ELS Component Schedules

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3.3.5 Environmental Control System

Figure 3-6 shows that the ECS is installed as a complete system in AFRM's 009 and 011. The system is not scheduled for completion of qualification until shortly after the launch of AFRM 011. Launch constraints will be required for the two flights of this system.

3.3.6 Electrical Power System

A study of Figure 3-7 shows that the fuel cells are not flown until AFRM 009 and that the complete system will be qualified shortly prior to the flight of B-18. Launch constraints will not be required for AFRM's 009 and 011 for the EPS.

3.3.7 Guidance and Navigation System

Figure 3-8 shows that the G&N system will be qualified for use for the first manned flight. Launch constraints, however, will be required for AFRM 009 for that portion of the G&N system installed on that flight vehicle.

3.3.8 Stabilization and Control System

Figure 3-9 shows that the SCS will not be qualified for use on any of the planned flights. Launch constraints will be required for B-22, AFRM 009, and AFRM 011 for the SCS.

3.3.9 Communications and Instrumentation Systems

Figure 3-10 shows qualified systems available for planned flight usage.

3.3.10 Life Systems

Figure 3-11 indicates the initial usage of the life systems on AFRM 011 with the system having completed qualification prior to that date.

3.3.11 Thermal Protection System

The information contained in Figure 3-12 reveals that the thermal protection systems is to be included on only AFRM's 009 and 011 with system qualification being accomplished considerably prior to that time.

3.3.12 Structure

Figure 3-13 portrays the usage of some portion of the primary structure on all flight vehicles. Structural development will be accomplished prior

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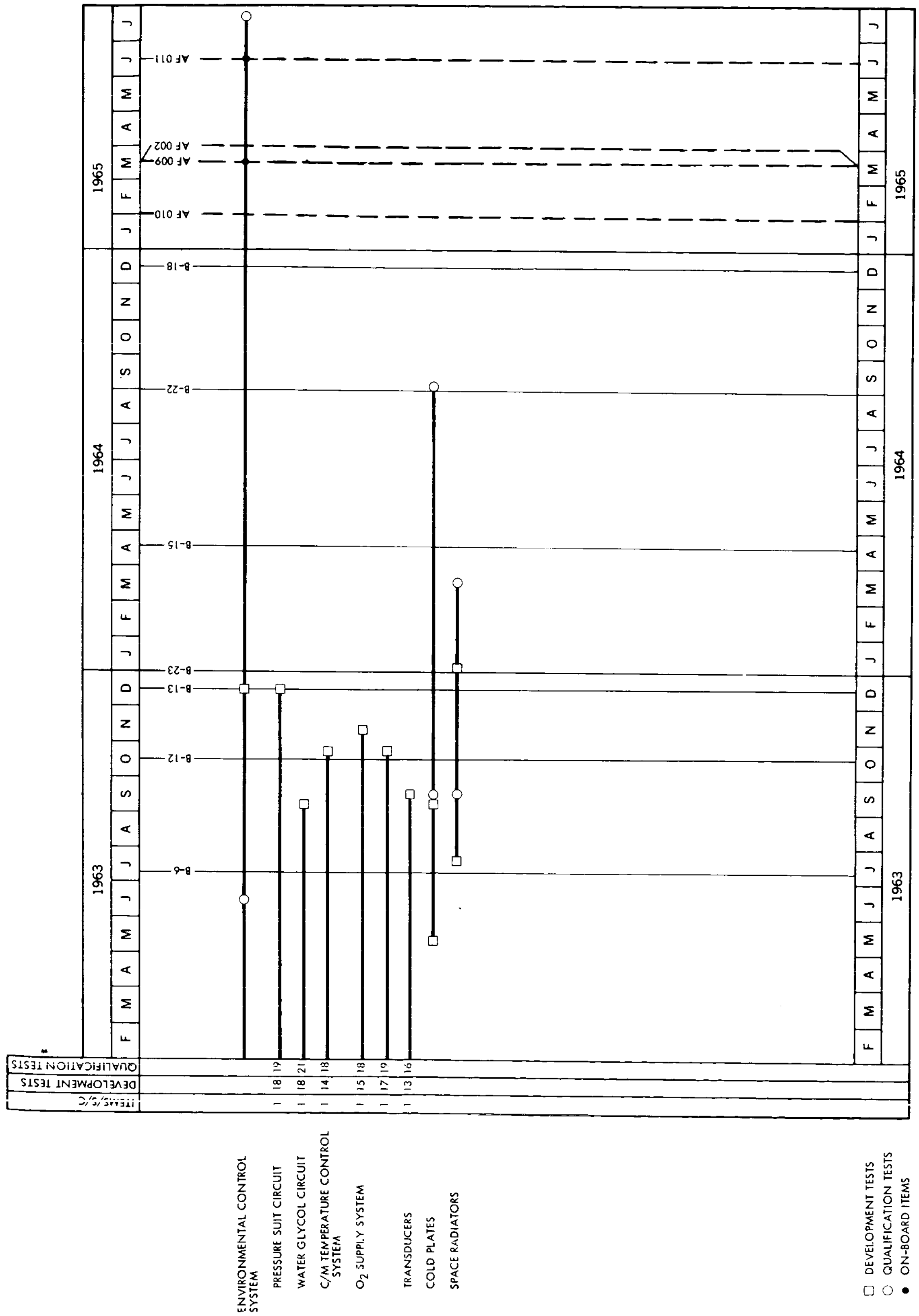

Figure 3-6. ECS Schedule Constraints

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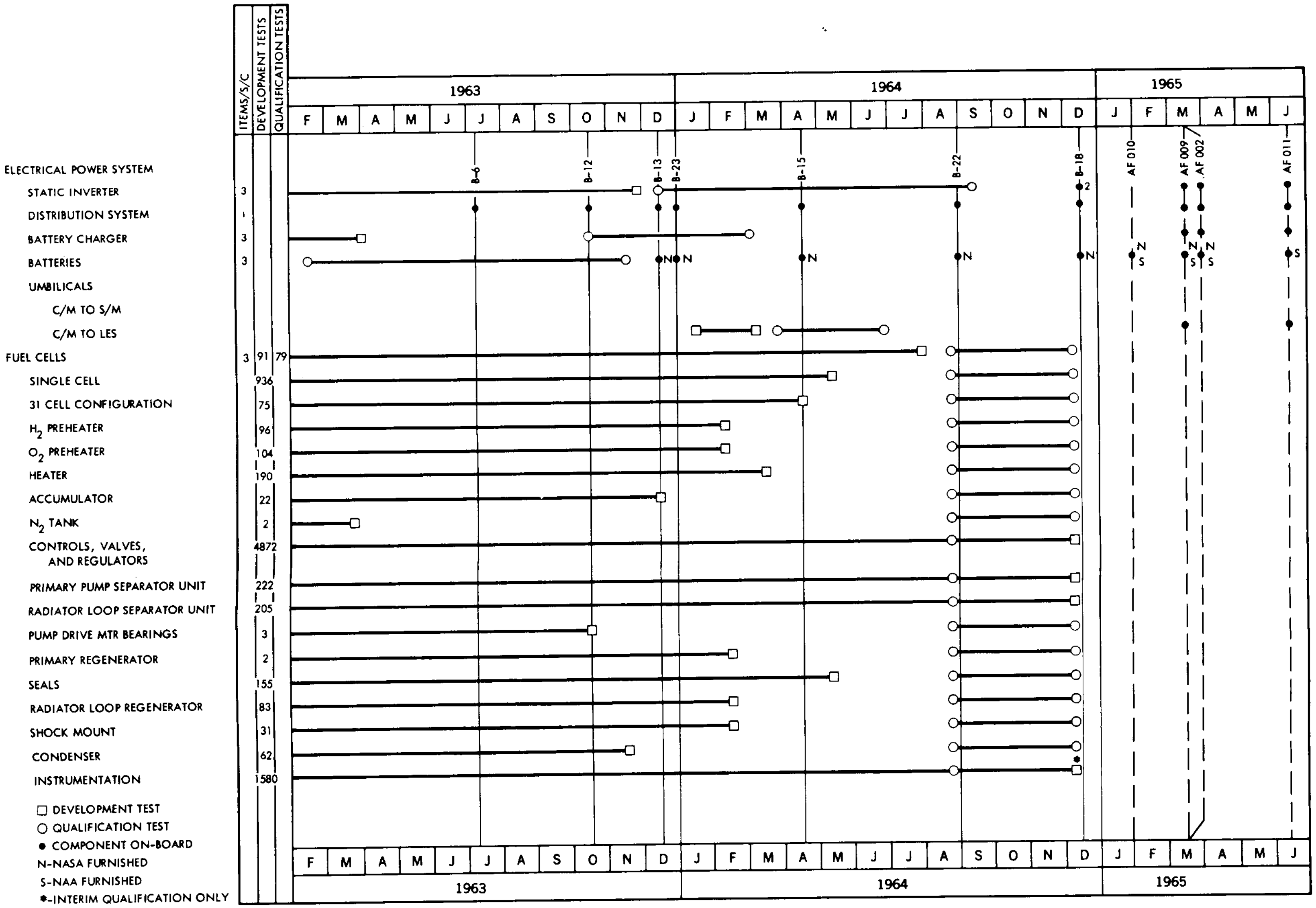

Figure 3-7. EPS Component Schedules

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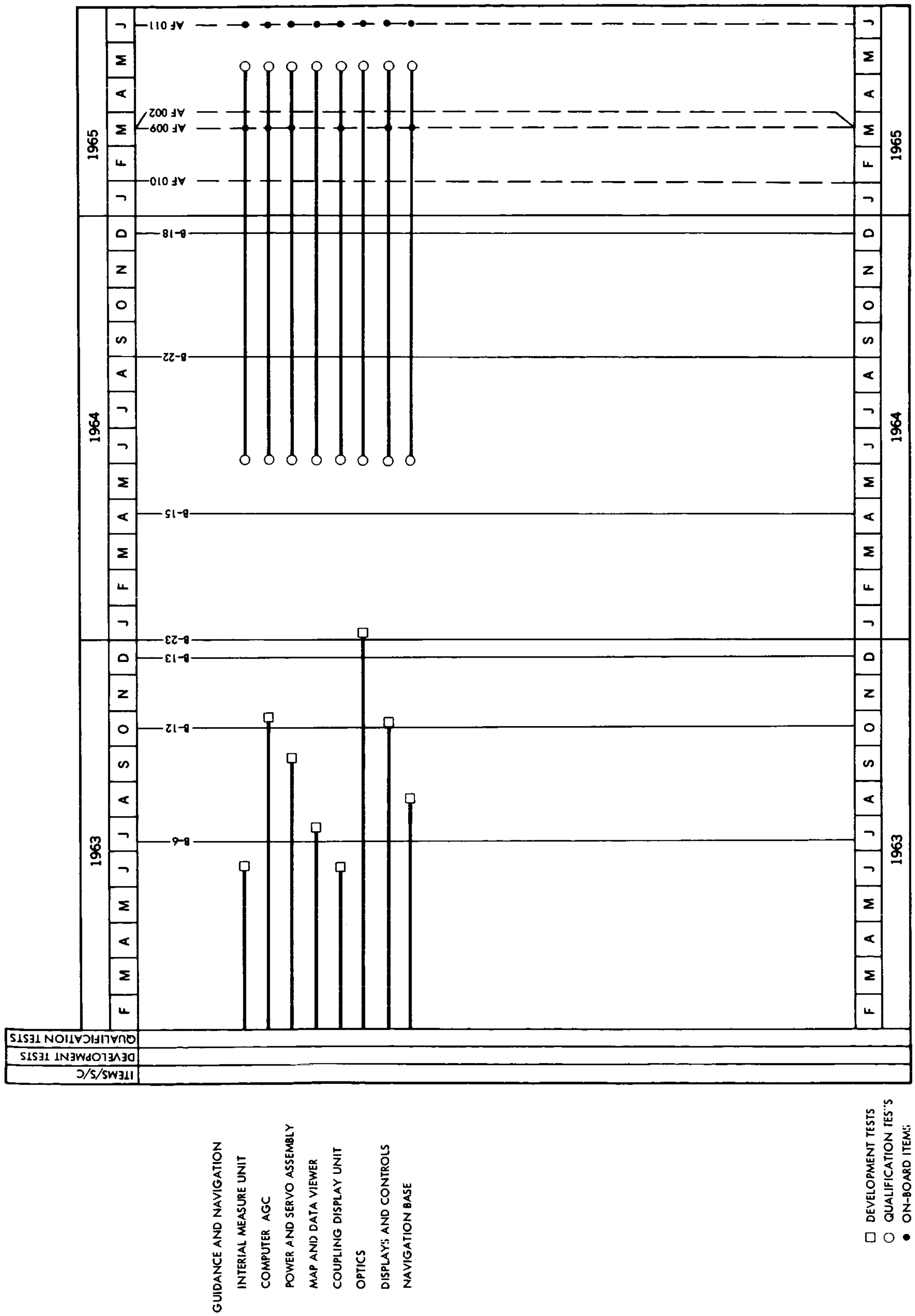

Figure 3-8. Guidance and Navigation System Component Schedules

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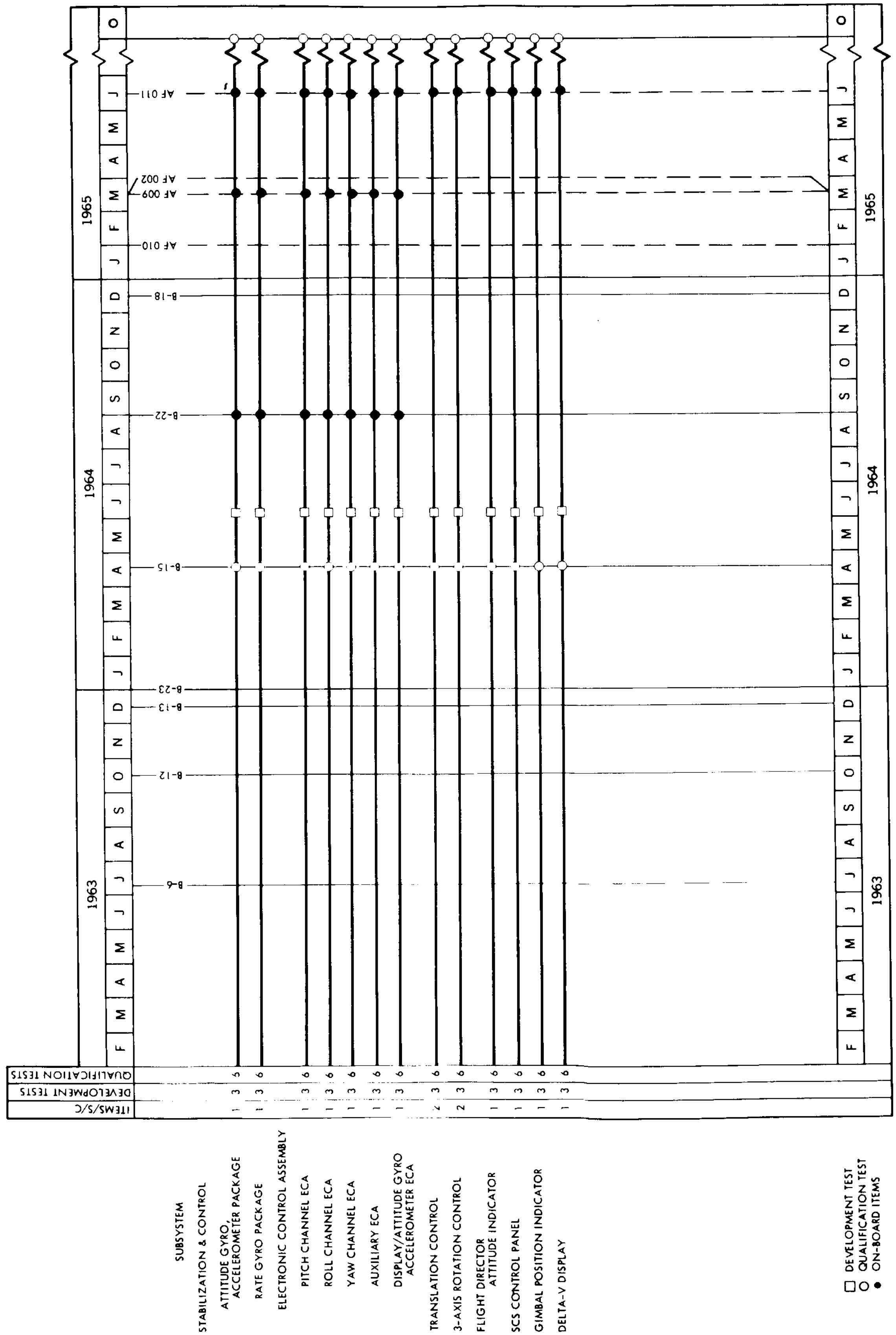

Figure 3-9. SCS Component Schedules

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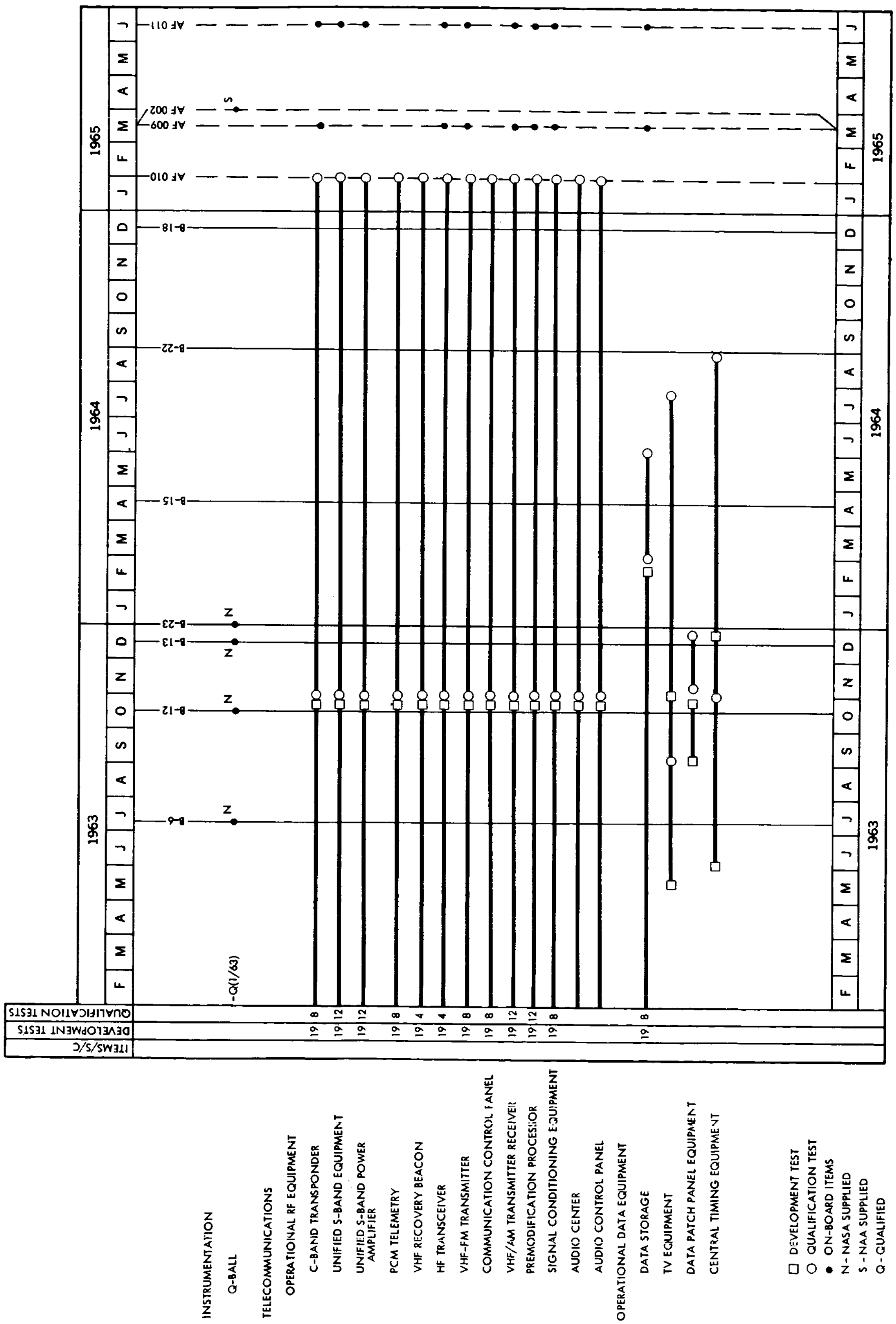

Figure 3-10. C&IS Component Schedules (Sheet 1 of 2)

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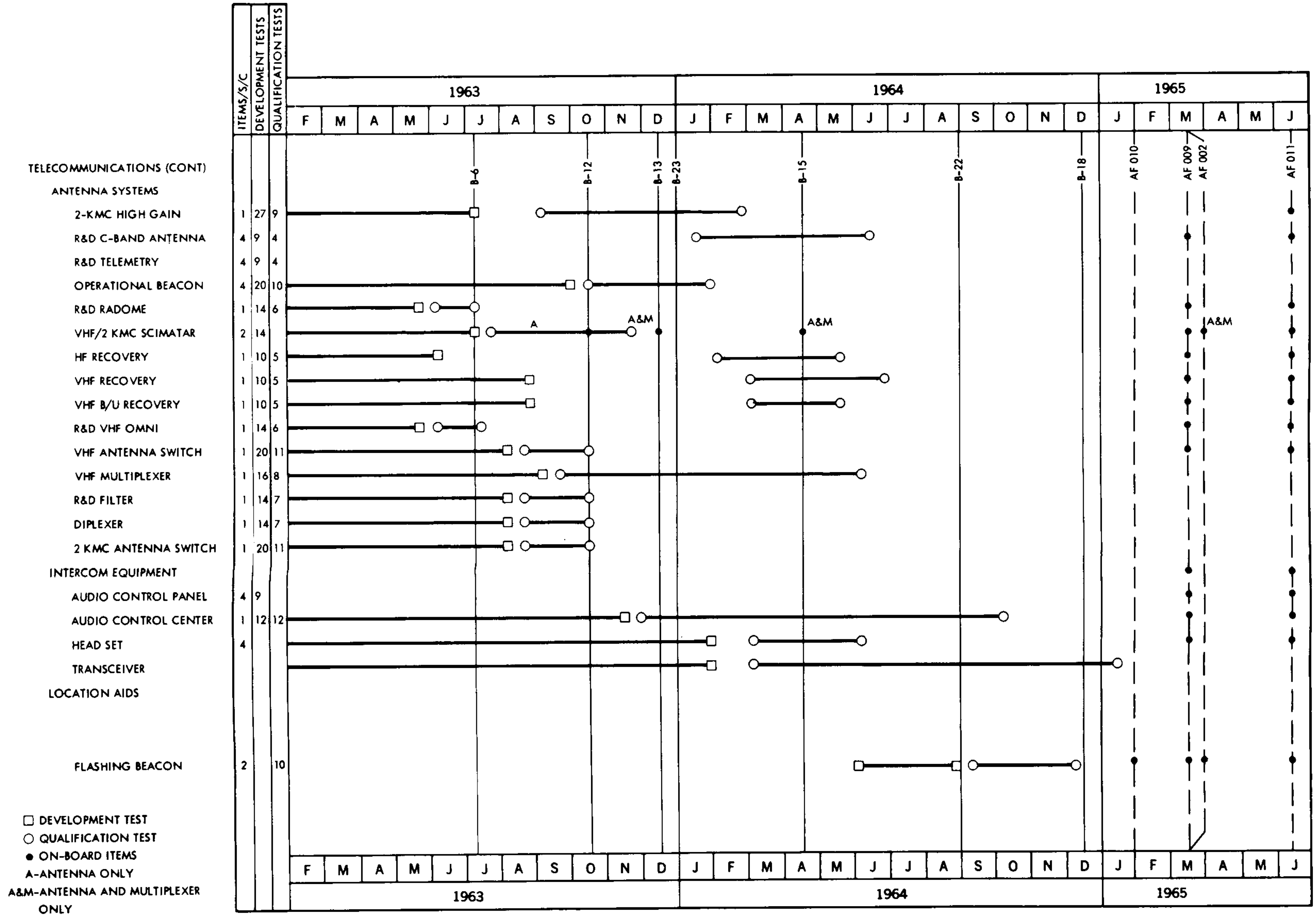

Figure 3-10. C&IS Component Schedule (Sheet 2 of 2)

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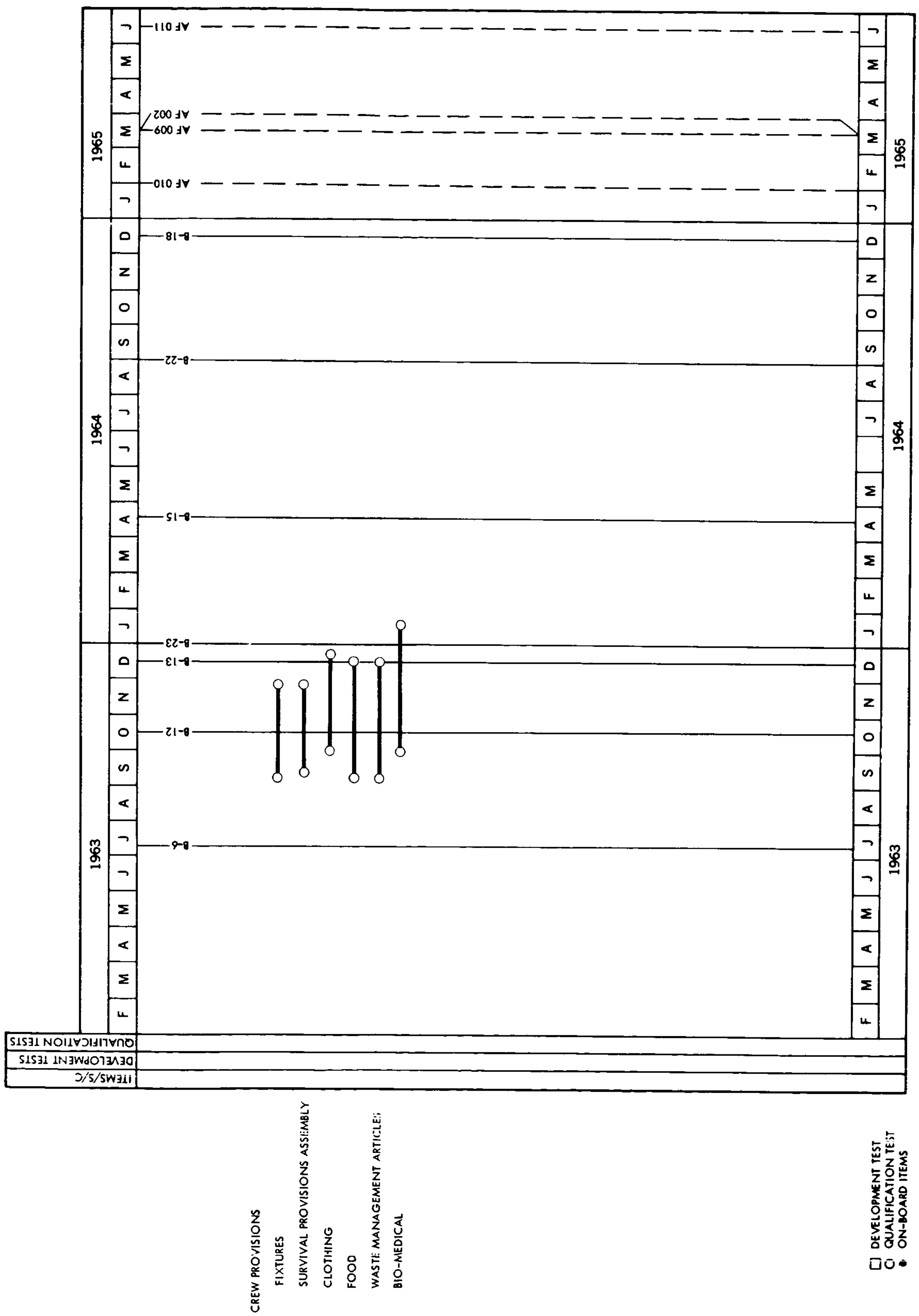

Figure 3-11. LS Component Schedules

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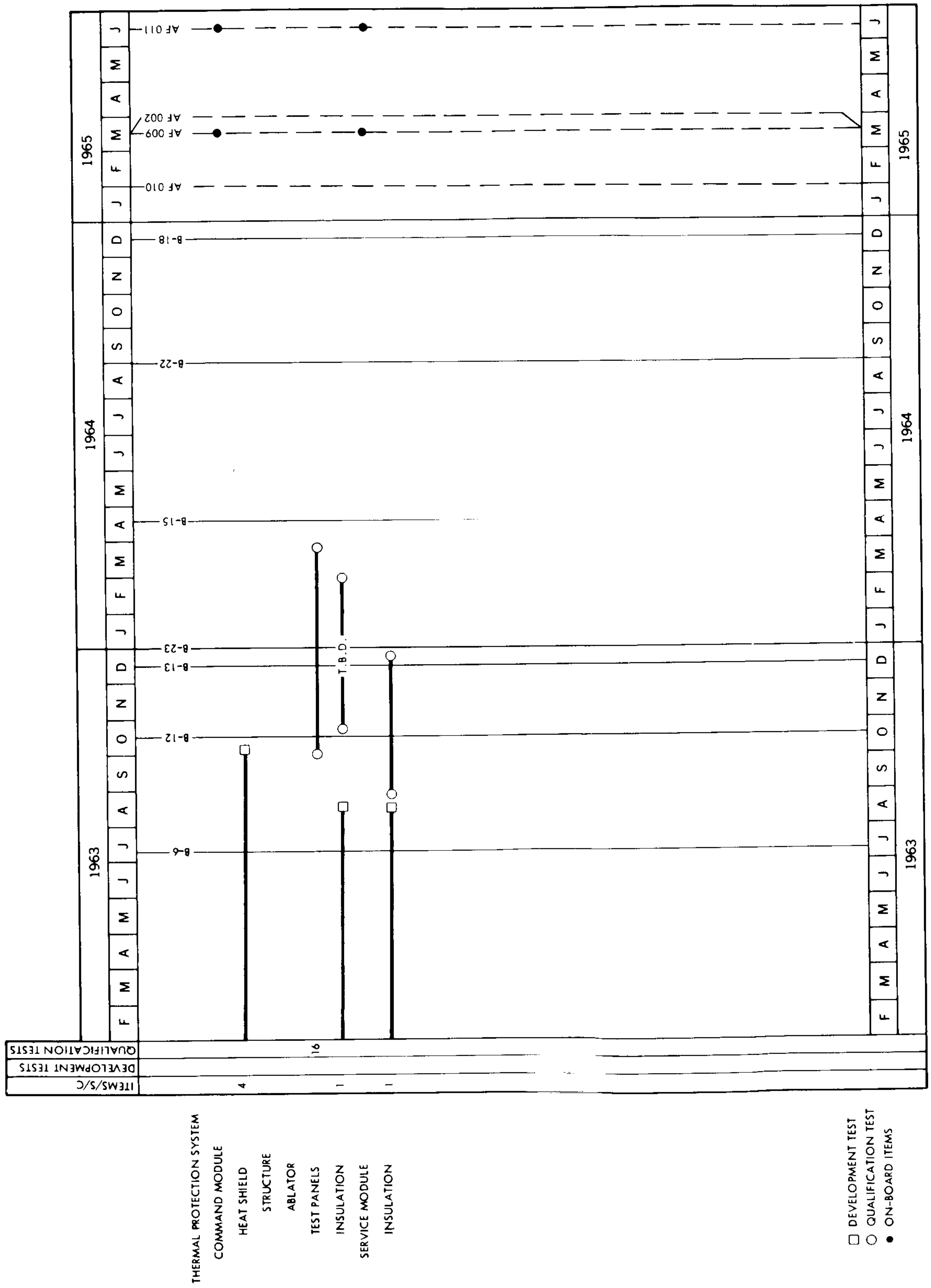

Figure 3-12. Thermal Protection System Component Schedules

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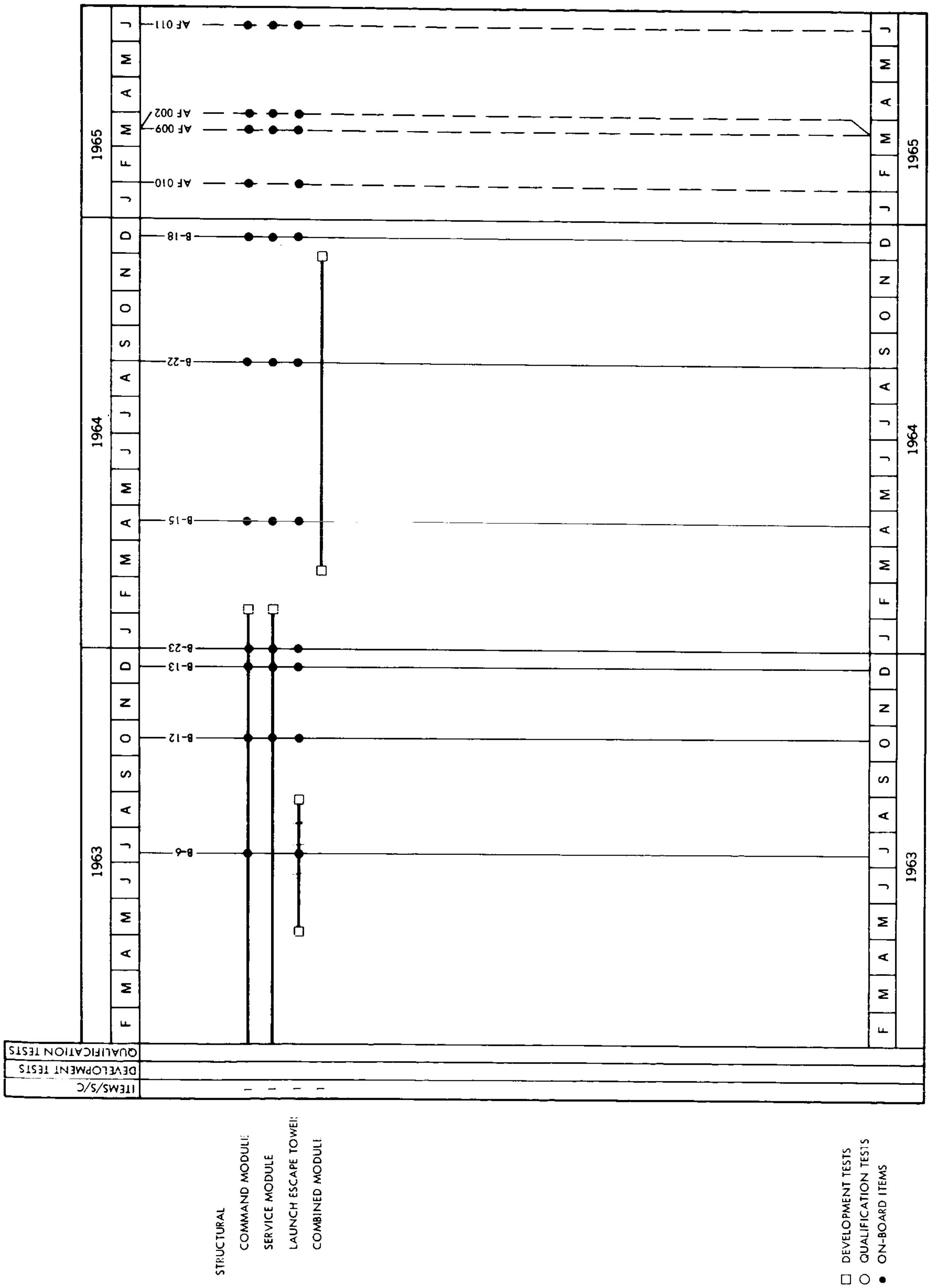

Figure 3-13. Structural Test Schedule

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to B-18. A test objective of B-18 is to qualify the physical and flight compatibility of the launch vehicle and spacecraft for manned flight.

3.3.13 Mechanical Systems

The mechanical systems, as included in Figure 3-14, will be used in all flight vehicles after B-15. Although launch constraints will be required for B-6, -12 and -23, all components of the system will have completed qualification prior to the flight of B-22.

3.3.14 Electronic Systems

Figure 3-15 shows that launch constraints will not be required for the planned flight usage of electronic systems.

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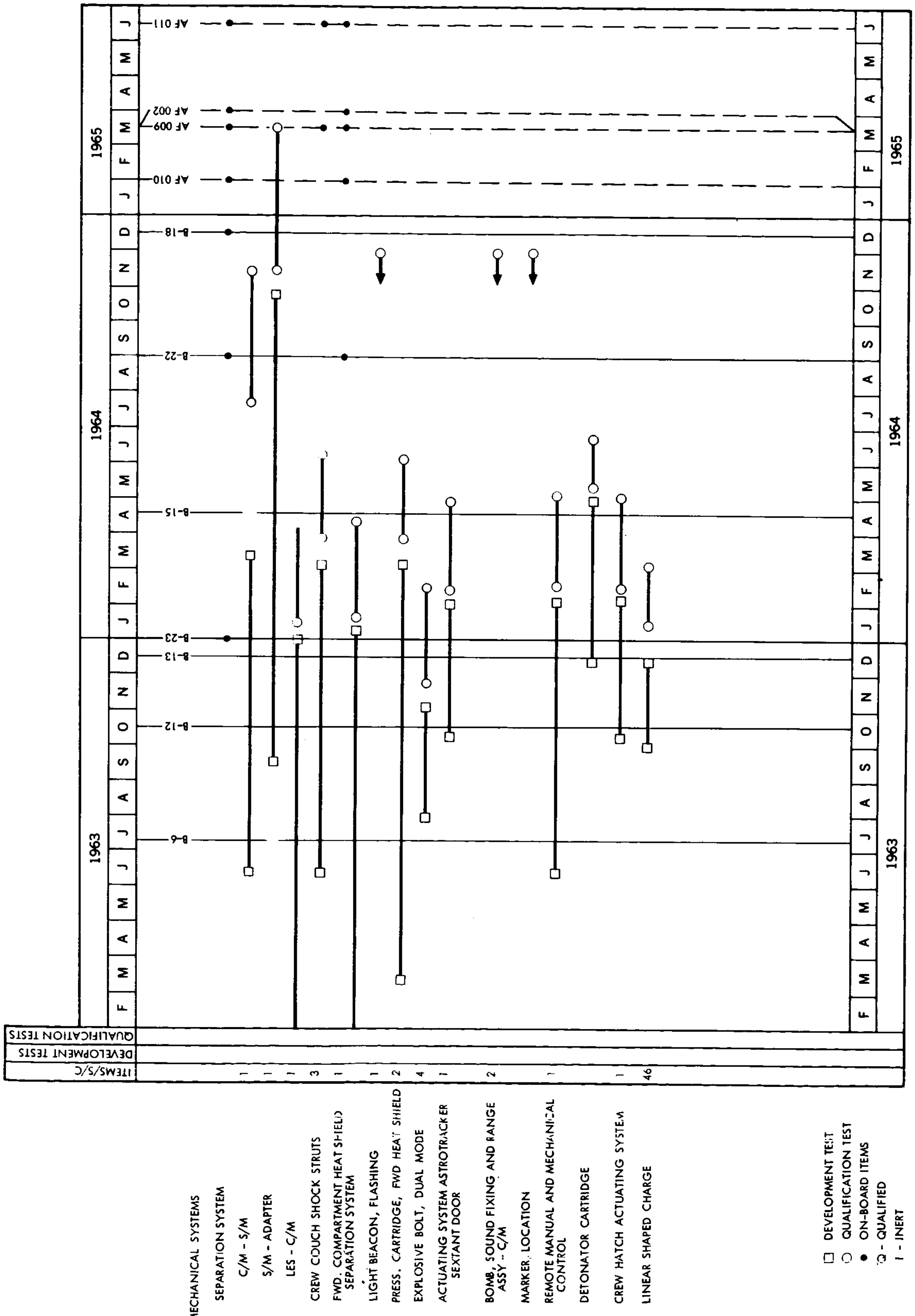

Figure 3-14. Mechanical Systems and Component Schedules

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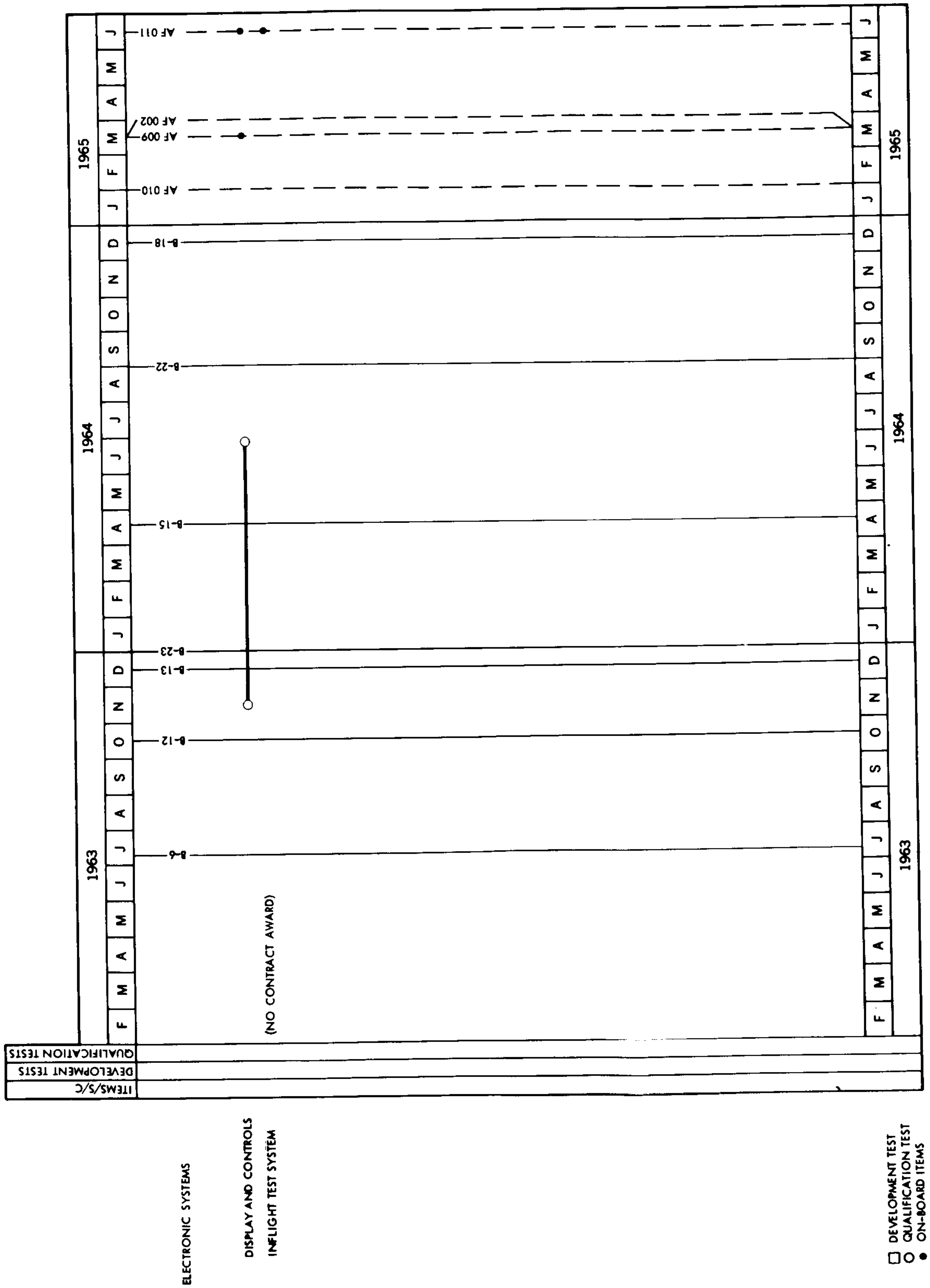

Figure 3-15. Electronic Systems Component Schedules

4.0 ENVIRONMENTAL CRITERIA

4.1 ENVIRONMENTAL CONSIDERATIONS

Section 4 defines the applicable Apollo program environmental criteria. These are defined in their order of occurrence, that is by prior-to-launch (prelaunch) phases and by mission phases, in Section 4.1. Section 4.2 describes in detail the specific environments and levels of environmental severity to be encountered during each of the prelaunch and mission phases.

The environmental criteria for the Apollo project have been separated into two major areas for consideration and treatment–the prior-to-launch environments and the post-launch or in-flight environments.

4.1.1 Prelaunch Environments

Figure 4-1 displays the environments to be encountered from completion of equipment manufacture to the actual point of launch. Specific environments are listed for each phase. The extent of equipment testing is indicated by phase.

4.1.2 LOR Mission Environments

Figure 4-2 displays the mission environments as they occur successively by phase. Generally, they exist as combined environments and are treated accordingly in the specific subsystem qualification test plans of Section 5. The combined environments to be employed during the mission-life tests are shown and related to the pertinent phase.

4.1.3 GSE Environments

Table 4-1 lists the environments anticipated for the GSE equipment. These items of equipment have been classified into two categories relevant to the imposed environments, sheltered and unsheltered. The environments to be imposed on them are tailored to fit these groups.

4.2 ENVIRONMENTAL LEVELS

This section delineates the levels of the different environments anticipated for the Apollo systems by program and mission phase.

Table 4-1. Ground Support Equipment Environments

Unsheltered Equipment	Sheltered Equipment
High temperature	Operating temperature
Low temperature	Humidity
Low pressure	Vibration
Humidity	Shock
Vibration	Electromagnetic interference
Shock	
Salt fog	
Sand and dust	
Operating temperature	
Electromagnetic interference	

4.2.1 Prelaunch Environmental Levels Anticipated

Some of the environments anticipated for the prior-to-launch phases will be considered only as a design requirement and will not be imposed during the testing phase. This is based on the premise that the packaging and/or shipping medium will compensate for the difference between the anticipated environment of paragraph 4.2 and the test levels of paragraph 4.3.

4.2.2 Transportation, Ground Handling, and Storage

These criteria represent the natural and induced environmental design criteria associated with transportation, ground handling, and storage. However, when a severe cost penalty or airborne weight penalty is imposed by one or more of these environmental criteria upon the design of Apollo equipment, the deviation from these criteria will be incorporated into the detail equipment specification. Shipping packages or special handling and transportation or storage provisions will be developed to prevent exposure to environmental conditions in excess of those which the equipment item is capable of withstanding.

MANUFACTURING COMPLETE → S&ID ACCEPTANCE → PARTS FROM CONTRACTOR → S&ID RECEIVING INSPECTION AND TEST → TEST PREPARATION

ACCEPTANCE TESTS (SOURCE INSPECTION)	PACKAGING AND STORAGE	SHIP TO S&ID	STORAGE	INSTALLATION	INDIVIDUAL SYSTEM CHECKOUT
ENVIRONMENTS PRODUCTION AMBIENT CONDITIONS	ENVIRONMENTS SHOCK TEMPERATURE HIGH AND LOW HUMIDITY	ENVIRONMENTS SHOCK TEMPERATURE HIGH AND LOW HUMIDITY VIBRATION	ENVIRONMENTS SHOCK TEMPERATURE HIGH AND LOW HUMIDITY	ENVIRONMENTS PRODUCTION AMBIENT CONDITIONS	ENVIRONMENTS PRODUCTION AMBIENT CONDITIONS
					EXTENT OF TEST TESTED TO SUBSYSTEM PERFORMANCE SPECIFICATIONS - WI ALL APPLICABLE VARIATIONS OF INPU AND LOADS

FOLD-OUT #1

SHIP TO AMR → MATE C/M AND S/M → STATIC FIRING OF RCS → SPS → MOVE TO VACUUM CHAMBER → MOVE TO LAUNCH AREA

S	INTEGRATED SYSTEMS CHECKOUT	INSPECTION AND MODIFICATION	INDIVIDUAL SYSTEMS CHECKOUT	COMBINED SYSTEMS CHECKOUT AND PRE-FIRING CHECK	FUEL CELL AND HIGH PRESSURE CHECKS	POST STATIC FIRING CHECKS	VACUUM CHAMBER CHECKS	INTEGRATED SYSTEMS CHECKOUT-COUNTDOWN AND LAUNCH
	ENVIRONMENTS PRODUCTION AMBIENT CONDITIONS	ENVIRONMENTS SHOCK TEMPERATURE HUMIDITY VIBRATION RAIN SALT SPRAY SAND AND DUST FUNGUS	ENVIRONMENTS PRODUCTION AMBIENT CONDITIONS WHICH WILL INCLUDE HIGH HUMIDITY	ENVIRONMENTS PRODUCTION AMBIENT CONDITIONS WHICH WILL INCLUDE HIGH HUMIDITY	ENVIRONMENTS PRODUCTION AMBIENT CONDITIONS WHICH WILL INCLUDE HIGH HUMIDITY	ENVIRONMENTS TEMPERATURE HUMIDITY VIBRATION RAIN SALT SPRAY SAND AND DUST FUNGUS ACOUSTICS	ENVIRONMENTS ALTITUDE	ENVIRONMENTS TEMPERATURE HUMIDITY SALT SPRAY FUNGUS RAIN
TH TS	EXTENT OF TEST TESTED TO VEHICLE PERFORMANCE SPECIFICATIONS WITH ALL SUBSYSTEMS CONNECTED		EXTENT OF TEST TESTED TO SUBSYSTEM PERFORMANCE SPECS WITH ALL APPLICABLE VARIATIONS OF INPUTS AND LOADS	EXTENT OF TEST TESTED TO VEHICLE PERFORMANCE SPECIFICATIONS WITH ALL SUBSYSTEMS INTERCONNECTED	EXTENT OF TEST HIGH-PRESSURE TEST FOR CHECKING CONNECTIONS, OPERATION OF FUEL CELL AND A CHECK OF POWER DISTRIBUTION PER SUBSYSTEM SPECS	EXTENT OF TEST ALL SYSTEMS OPERATING TO VEHICLE SPECIFICATIONS	EXTENT OF TEST LEAKAGE TEST ONLY	EXTENT OF TEST ALL SYSTEM CHECKOUT TO VEHICLE SPECIFICATIONS

FOLD-OUT #2

Figure 4-1. Apollo S/C Prelaunch Environments

[redacted]

[redacted]

LAUNCH — JETTISON LET — SECOND-STAGE BURNOUT — LEM ACTUATION — LEM EJECT — LEM LANDI

FIRST-STAGE BOOST	SECOND-STAGE BOOST	EARTH PARK ORBIT	TRANSLUNAR INJECTION	LEM TRANSFER	TRANSLUNAR COAST	LUNAR ORBIT INJECTION	LUNAR ORBIT	
ANTICIPATED ENVIRONMENTS ACOUSTIC VIBRATION ACCELERATION	ACOUSTIC VIBRATION ACCELERATION	VACUUM HIGH TEMPERATURE LOW TEMPERATURE ZERO g	ACOUSTIC VIBRATION ACCELERATION VACUUM HIGH TEMPERATURE LOW TEMPERATURE ZERO g	VACUUM HIGH TEMPERATURE LOW TEMPERATURE ZERO g SHOCK	VACUUM HIGH TEMPERATURE LOW TEMPERATURE ZERO g	ACOUSTIC VIBRATION ACCELERATION VACUUM HIGH TEMPERATURE LOW TEMPERATURE ZERO g	VACUUM HIGH TEMPERATURE LOW TEMPERATURE ZERO g	VACUUM HIGH TEMPERATURE LOW TEMPERATURE ZERO g
SIMULATED MISSION LIFE ENVIRONMENTS VIBRATION AND ACOUSTICS VIBRATION AND ACCELERATION	VIBRATION AND ACOUSTICS VIBRATION AND ACCELERATION	VIBRATION AND ACOUSTICS VIBRATION AND ACCELERATION	VIBRATION AND ACOUSTICS VIBRATION AND ACCELERATION	VIBRATION AND ACOUSTICS VIBRATION AND ACCELERATION	VIBRATION AND ACOUSTICS VIBRATION AND ACCELERATION	VIBRATION AND ACOUSTICS VIBRATION AND ACCELERATION	VIBRATION AND ACOUSTICS VIBRATION AND ACCELERATION	VIBRATION AND ACOUSTICS VIBRATION AND ACCELERATION

FOLD-OUT #1

NG	LEM LAUNCH	BEGIN LUNAR ORBIT			4000 MILES TO ENTRY	BEGIN ENTRY	LANDING

	LEM ASCENT	APOLLO-LEM RENDEZVOUS	TRANSEARTH INJECTION	TRANSEARTH COAST	ENTRY	RECOVERY
		VIBRATION	VIBRATION ACCELERATION		VIBRATION DECELERATION	
VACUUM HIGH TEMPERATURE LOW TEMPERATURE ZERO g	VACUUM HIGH TEMPERATURE LOW TEMPERATURE ZERO g	VACUUM HIGH TEMPERATURE LOW TEMPERATURE ZERO g SHOCK	VACUUM HIGH TEMPERATURE LOW TEMPERATURE ZERO g	VACUUM HIGH TEMPERATURE LOW TEMPERATURE ZERO g	HIGH TEMPERATURE SHOCK	
VIBRATION AND ACOUSTICS VIBRATION AND ACCELERATION	VIBRATION AND ACOUSTICS VIBRATION AND ACCELERATION	VIBRATION AND ACOUSTICS VIBRATION AND ACCELERATION	VIBRATION AND ACOUSTICS VIBRATION AND ACCELERATION	VACUUM HIGH TEMPERATURE VACUUM LOW TEMPERATURE	SHOCK VERY HIGH TEMPERATURE ACCELERATION	

Figure 4-2. Apollo LOR Mission Phases Environments

FOLD-OUT #2

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4.2.2.1 Natural Environments

1. Temperature

   a. Air Transportation. -20 F minimum to +140 F maximum for 8 hours

   b. Ground Transportation. -20 F minimum to +140 F maximum for 2 weeks

   c. Storage. +25 F minimum to +105 F maximum for 3 years

   d. Launch Base Area. +25 F minimum to +105 F maximum air temperature for 30 days, plus solar radiation of 360 Btu/ft^2/hr for a 6-hour period each day

2. Altitude

   a. Air Transportation. Up to 35,000 feet for 8 hours

   b. Storage. Up to 6000 feet for 3 years

3. Humidity. 95 ± 5 percent relative humidity, including conditions wherein condensation takes place in the form of water

4. Sunshine. Solar radiation at 360 Btu/ft^2/hour for 6 hours per day for 2 weeks

5. Rain. Up to 0.6 inches/hour for 12 hours

6. Sand and Dust. As encountered in desert and ocean beach areas, equivalent to 140 mesh silica flour, with particle velocity up to 500 feet per minute

7. Fungus. As experienced in tropical climates. The use of non-nutrient materials is recommended wherever possible. Where materials which are nutrients for fungi are used, such materials will be treated with an approved fungicidal agent. Fungus resistant materials are as defined in Specification MIL-E-5400.

8. Salt Spray. Salt atmosphere as encountered in coastal areas, the effect of which is simulated by 20 percent salt solution for 50 hours.

9. Ozone. Up to 3 years exposure to 0.05 parts/million concentration

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[redacted] [redacted]

10. Ground Winds. These ground wind criteria consist of a description of Cape Canaveral wind data for the height intervals of 10 to 400 feet.

a. Free Standing. The design peak wind speeds for structural loading studies of the Saturn and Nova-type vehicles (fueled or unfueled) provides a free-standing capability 99.9 percent of the time during the strongest wind month at Cape Canaveral. The gradient to provide this capability is presented in the following tabulation.

Height	Steady State Wind	Peak Wind *
10	23.0 knots	32.2 knots
30	28.8	40.3
60	33.6	47.0
100	37.5	52.5
200	42.6	59.6
300	46.0	64.4
400	48.3	67.6

*Gust Characteristics. For the effects of gusts, a linear build up from the steady-state winds to the peak winds will be assumed. The period of this build up and decay will be taken as 4 seconds for all height levels, i.e., build up of 2 seconds and 2 seconds for decay to steady-state wind speed.

b. Storm Conditions. The 99.9-percent peak wind speeds may be exceeded during severe thunderstorm or hurricane conditions at Cape Canaveral. During such periods, the vehicle must be placed in a service structure, shelter, or tied in such a manner that wind loading conditions greater than those for the 99.9-percent winds will not be experienced by the vehicle. The vehicle protective structures should be designed to withstand wind loads resulting from a probable maximum wind speed of 108 knots while containing the vehicle. Wind speeds apply from 10 to 400 feet heights.

4.2.2.2 Induced Environments

1. Shock. As experienced in each of three mutually perpendicular axes

Equipment Weight	Shock Level	Time
Less than 250 lbs	30 g	11 ± 1 milliseconds
250 lbs to 500 lbs	24 g	11 ± 1 milliseconds
500 lbs to 1000 lbs	21 g	11 ± 1 milliseconds
Over 1000 lbs	18 g	11 ± 1 milliseconds

2. Vibration. As experienced in each of three mutually perpendicular axes

Equipment Weight	5 to 27.5 cps	27.5 to 52 cps	52 to 500 cps
Less than 50 lbs	±1.56 g	0.043 in double amplitude	±6.0 g
50 to 1000 lbs	±1.30 g	0.036 in double amplitude	±5.0 g
Over 1000 lbs	±1.04 g	0.029 in double amplitude	±3.33 g

4.2.3 Post-Launch Environmental Levels

The post-launch environments are presented as a function of mission phase and spacecraft zone. Figure 4-3 illustrates the four zones of the spacecraft as a function of station number and function.

The vibration, acceleration, acoustic, and shock environments are shown in Table 4-2.

The temperature environments are described in Table 4-3.

Other environments, such as oxidation, hazardous gases, radiation, pressure, winds, humidity, radio interference, and meteoroids, are as follows:

1. Oxidation. Command module equipment will withstand an atmosphere of 100 percent oxygen at 5 psia for 400 hours (includes ground and flight time).

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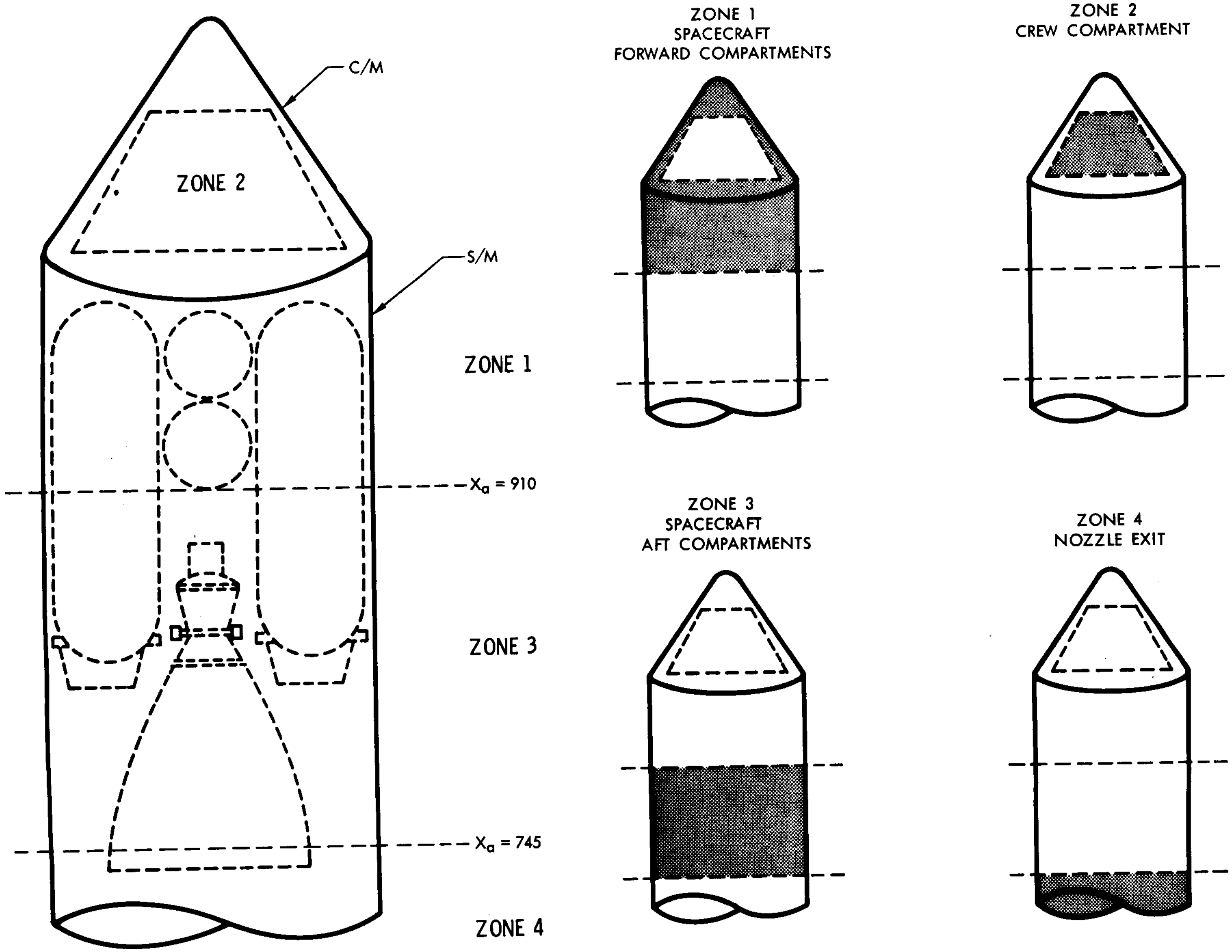

Figure 4-3. Spacecraft Zones, Location and Title

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Table 4-2. Flight Mission Phases (Sheet 1 of 8)

The vibration levels shown are for basic structure and packages rigidly mounted thereon. For rigidly mounted packages with weight in excess of 10 pounds, the vibration levels at frequencies above one-half of the first local response mode will diminish in accordance with the following: (g) $A_2 = A_1 (1.5 - 1/2\ 10g\ W_2)$ Where A_1 = noted vibration level, (g) A_2 = vibration level of package, (lb) W_2 = weight of package

All references to linear increases or decreases in environments are predicated on log - log scale relationships

	Criteria	Lift-Off	Boost	Earth Orbit	Trans-Lunar Injection and Coast	Lunar Orbit	Trans-Earth Injection and Coast	Earth Entry	Earth Landing	Pad Abort	Max q Abort	Very High Altitude Abort
COMMAND MODULE and SERVICE MODULE Forward of Station Xa = 910	ACOUSTICS (Sound Press Level-db-Ref. 0002 dynes per cm^2) Frequency / External / Internal 4.7 - 9.4 / 155 db / 145 db 9.4 - 18.8 / 163 / 153 18.8 - 37.5 / 164 / 150 37.5 - 75 / 161 / 150 75 - 150 / 158 / 147 150 - 300 / 155 / 145 300 - 600 / 150 / 140 600 - 1200 / 145 / 132 1200 - 2400 / 139 / 125 2400 - 4800 / 133 / 120 4800 - 9600 / 129 / 116 Overall / 168.5 / 158	←180 sec.→						120 sec.				
	RANDOM VIBRATION Linear increase from 0.007 g^2/cps at 5 cps to 0.122 g^2/cps at 50 cps 0.122 g^2/cps from 50 to 150 cps with a linear decrease to 0.007 g^2/cps at 2000 cps.	←180 sec.→						120 sec.				
	PEAK VIBRATION ACCELERATION Linear increase from 0.3 g at 5 cps to 10 g at 100 cps. 10 g from 100 to 2000 cps.	←180 sec.→						120 sec.				
	RANDOM VIBRATION Linear increase from 0.0075 g^2/cps at 5 cps to 0.052 g^2/cps at 100 cps. 0.052 g^2/cps from 100 to 550 cps with a linear decrease to 0.035 g^2/cps at 2000 cps.				40 sec.	433 sec.	176 sec.					410 sec.
	PEAK VIBRATION Linear increase from 0.16 g at 5 cps to 5.2 g from 300 to 2000 cps.											

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Table 4-2. Flight Mission Phases (Sheet 2 of 8)

	Criteria			Lift-Off	Boost	Earth Orbit	Trans-Lunar Injection and Coast	Lunar Orbit	Trans-Earth Injection and Coast	Earth Entry	Earth Landing	Pad Abort	Max q Abort	Very High Altitude Abort
	ACOUSTICS													
	Frequency	External	db											
	4.7 - 9.4	154	144									8 sec.	30 sec	
	9.4 - 18.8	162	152											
	18.8 - 37.5	165	151											
	37.5 - 75	161	150											
	75 - 150	161	150											
	150 - 300	159	149											
	300 - 600	157	144											
	600 - 1200	158	145											
	1200 - 2400	159	147											
	2400 - 4800	157	144											
	4800 - 9600	155	142											
	Overall	171	158											
	RANDOM VIBRATION													
	Linear increase from 0.006 g^2/cps at 5 cps to 0.155 g^2/cps at 60 cps. 0.155 g^2/cps from 60 to 500 cps with a linear decrease to 0.035 g^2/cps at 2000 cps.													
	PEAK VIBRATION ACCELERATION													
	Linear increase from 0.3 g at 5 cps to 2 g at 15 cps. Another linear increase to 11 g at 100 cps. Another linear increase to 25 g at 1000 cps. 25 g from 1000 to 2000 cps.													
Crew Compartment	ACOUSTICS													
	Frequency	db												
	4.7 - 9.4	131		← 180 sec.	→					120 sec.				
	9.4 - 18.8	132												
	18.8 - 37.5	133												
	37.5 - 75	133												
	75 - 150	131												
	150 - 300	127												
	300 - 600	122												
	600 - 1200	116												
	1200 - 2400	109												
	2400 - 4800	101												
	4800 - 9600	95												
	Overall	140												

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Table 4-2. Flight Mission Phases (Sheet 3 of 8)

	Criteria	Lift-Off	Boost	Earth Orbit	Trans-Lunar Injection and Coast	Lunar Orbit	Trans-Earth Injection and Coast	Earth Entry	Earth Landing	Pad Abort	Max q Abort	Very High Altitude Abort
	RANDOM VIBRATION Linear increase from 0.0063 g^2/cps at 5 cps to 0.095 g^2/cps at 50 cps. 0.095 g^2/cps from 50 to 150 cps with a linear decrease to 0.0035 g^2/cps at 2000 cps.	←180 sec.→						120 sec.				
	PEAK VIBRATION ACCELERATION Linear increase from 0.3 g at 5 cps to 8.5 at 100 cps. 8.5 g from 100 to 300 cps with a linear decrease to 5 g at 2000 cps.	←180 sec.→						120 sec.				
	RANDOM VIBRATION Linear increase from 0.007 g^2/cps at 5 cps to 0.040 g^2/cps at 100 cps. 0.040 g^2/cps from 100 to 200 cps with a linear decrease to 0.015 g^2/cps at 2000 cps.				40 sec.	433 sec.	176 sec.					410 sec.
	PEAK VIBRATION Linear increase from 0.16 g at 5 cps to 3.5 g at 300 cps, with a linear decrease to 2.5 g at 2000 cps.				40 sec.	433 sec.	176 sec.					410 sec.
	ACOUSTICS Frequency — db 4.7 - 9.4 — 131 9.4 - 18.8 — 132 18.8 - 37.5 — 135 37.5 - 75 — 134 75 - 150 — 134 150 - 300 — 131 300 - 600 — 129 600 - 1200 — 130 1200 - 2400 — 129 2400 - 4800 — 126 4800 - 9600 — 122 Overall — 142									8 sec.	30 sec	

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Table 4-2. Flight Mission Phases (Sheet 4 of 8)

	Criteria	Lift-Off	Boost	Earth Orbit	Trans-Lunar Injection and Coast	Lunar Orbit	Trans-Earth Injection and Coast	Earth Entry	Earth Landing	Pad Abort	Max q Abort	Very High Altitude Abort
Service Module From STA Xa = 910 to STA Xa = 745 of the Adapter.	RANDOM VIBRATION Linear increase from 0.006 g^2/cps at 5 cps to 0.13 g^2/cps at 60 cps. 0.13 g^2/cps from 60 to 200 cps with a linear decrease to 0.006 g^2/cps at 2000 cps.									8 sec.	30 sec.	
	PEAK VIBRATION ACCELERATION Linear increase from 0.3 g at 5 cps to 2 g at 15 cps. Another linear increase to 11 g at 100 cps. 11 g from 100 to 2000 cps.									8 sec.	30 sec.	
	ACOUSTICS Frequency — db 4.7 - 9.4 — 137 9.4 - 18.8 — 141 18.8 - 37.5 — 142 37.5 - 75 — 143 75 - 150 — 140 150 - 300 — 1 6 300 - 600 — 132 600 - 1200 — 122 1200 - 2400 — 119 2400 - 4800 — 113 4800 - 9600 — 107 Overall — 148	←180 sec.→						120 sec.				
	RANDOM VIBRATION Linear increase from 0.005 g^2/cps at 5 cps to 0.086 g^2/cps at 50 cps. 0.086 g^2/cps from 50 to 150 cps with a linear decrease to 0.005 g^2/cps at 2000 cps.	←180 sec.→						120 sec.				
	PEAK VIBRATION ACCELERATION Linear increase from 0.3 g at 5 cps to 7 g at 100 cps. 7 g from 100 to 2000 cps.	←180 sec.→						120 sec.				

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Table 4-2. Flight Mission Phases (Sheet 5 of 8)

	Criteria	Lift-Off	Boost	Earth Orbit	Trans-Lunar Injection and Coast	Lunar Orbit	Trans-Earth Injection and Coast	Earth Entry	Earth Landing	Pad Abort	Max q Abort	Very High Altitude Abort
Adapter AFT of Station Xa = 745	RANDOM VIBRATION Linear increase from 0.0086 g^2/cps at 5 cps to 0.070 g^2/cps at 100 cps. 0.070 g^2/cps from 100 to 550 cps, with a linear decrease to 0.050 g^2/cps at 2000 cps.				40 sec.	433 sec.	176 sec.					410 sec.
	PEAK VIBRATION ACCELERATION Linear increase from 0.23 g at 5 cps to 9 g at 400 cps. 9 g from 400 to 2000 cps.				40 sec.	433 sec.	176 sec.					410 sec.
	ACCUSTICS (see frequency table below)	←180 sec.→						120 sec.				
	RANDOM VIBRATION Linear increase from 0.004 g^2/cps at 5 cps to 0.025 g^2/cps at 50 cps. 0.025 g^2/cps from 50 to 150 cps, with a linear decrease to 0.005 g^2/cps at 2000 cps.	←180 sec.→						120 sec.				
	PEAK VIBRATION ACCELERATION Linear increase from 0.3 g at 5 cps to 5 g at 120 cps. 5 g from 120 to 2000 cps.	←180 sec.→						120 sec.				

Frequency	db
4.7 - 9.4	136
9.4 - 18.8	142
18.8 - 37.5	141
37.5 - 75	141
75 - 150	138
150 - 300	134
300 - 600	130
600 - 1200	123
1200 - 2400	116
2400 - 4800	110
4800 - 9600	104
Overall	147

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Table 4-2. Flight Mission Phases (Sheet 6 of 8)

	Criteria	Lift-Off	Boost	Earth Orbit	Trans-Lunar Injection and Coast	Lunar Orbit	Trans-Earth Injection and Coast	Earth Entry	Earth Landing	Pad Abort	Max q Abort	Very High Altitude Abort
COMMAND MODULE	ACCELERATION (Sustained) Applied along longitudinal axis of vehicle: Peak Acceleration - Total Duration -	7 g → 147 sec.						20 g 120 sec.		20 g 4 sec.	20 g 4 sec.	
	SHOCK 78 g This shock environment is expected to exist as a result of C/M structural response to an emergency earth landing condition. No hazard to crew. Only recovery aids, operate after.								Appli-cable			
SERVICE MODULE	ACCELERATION (Sustained) Applied along longitudinal axis of vehicle: Peak Acceleration - Total Duration -	7 g → 147 sec.										
CREW COMPARTMENT	ACOUSTICS For extended durations the overall noise levels resulting from all component noise sources shall not exceed 80 db (re 0.0002 dynes/cm^2) in the frequency range of 20 to 10,000 cps, nor produce a speech interference level (SIL) greater than 55 db. NOTE: SIL is defined as the arithmetic average of the acoustic levels in the octave bands with geometric mean frequencies of 400, 800, 1600, and 3200 cps.				40 sec.	433 sec.	176 sec.					
SPACECRAFT C/M AND S/M	Decking Dynamics Criteria: Not determined.											

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Table 4-2. Flight Mission Phases (Sheet 7 of 8)

	Criteria	Lift-Off	Boost	Earth Orbit	Trans-Lunar Injection and Coast	Lunar Orbit	Trans-Earth Injection and Coast	Earth Entry	Earth Landing	Pad Abort	Max q Abort	Very High Altitude Abort
COMMAND MODULE	EARTH LANDING IMPACT ACCELERATIONS											
	Crew Couch											
	x-x (eyeballs in) 26.6 g's-Crew capacity x-x (eyeballs out) 19.5 g's-Tumbling-Launch abort		z-z (eyeballs up) 15.0 g's - System Launch z-z (eyeballs down) 20.0 g's - System Launch y-y (eyeballs side) 15.0 g's - System Launch									
	Nominal for C/M											
	x-x (eyeballs in) 23.3 g's y-y (eyeballs side) 15.2 g's		z-z (eyeballs down) 19.3 g's z-z (eyeballs up) 10.1 g's									
	*Emergency for C/M - Internal Equip. Support											
	x-x (eyeballs in) 52.0 g's x-x (eyeballs out) 19.5 g's		z-z (eyeballs down) 46.8 g's z-z (eyeballs up) 34.6 g's y-y (eyeballs side) 34.6 g's									
Launch Escape System	RANDOM VIBRATION											
	Linear increase from 0.007 g^2/cps at 5 cps to 0.122 g^2/cps at 50 cps. 0.122 g^2/cps from 50 to 150 cps with a linear decrease to 0.007 g^2/cps at 2000 cps.	←115 sec.→									4 sec.	4 sec.
	PEAK VIBRATION ACCELERATION											
	Linear increase from 0.3 g at 5 cps to 10 g at 100 cps. 10 g from 100 to 2000 cps.	←115 sec.→									4 sec.	4 sec.

*Based upon: Deletion of earth landing impact attenuation system and no deployment of aft heat shield during earth landing; pitch orientation revised to -5° maximum variation about horizontal Y axis of C/M with respect to ground vertical.

Reference axes are parallel and perpendicular to the C/M. The above values are to be combined in an elliptical form to obtain off axis loading where critical. A factor of 1.5 is to be applied to above values to obtain ultimate loads.

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Table 4-2. Flight Mission Phases (Sheet 8 of 8)

	Criteria	Lift-Off	Boost	Earth Orbit	Trans-Lunar Injection and Coast	Lunar Orbit	Trans-Earth Injection and Coast	Earth Entry	Earth Landing	Pad Abort	Max q Abort	Very High Altitude Abort
	ACOUSTICS Octave Band Frequencies / LES Skirt Area / LES Cone Area 37.5 - 75 / 143 db / 137 db 75 - 150 / 144 / 136 150 - 300 / 145 / 132 300 - 600 / 147 / 134 600 - 1200 / 149 / 139 1200 - 2400 / 153 / 144 2400 - 4800 / 156 / 146 4800 - 9600 / 155 / 148 Overall / 160 / 153 SHOCK This criteria applies to the instruments mounted in the nose cone area only. The transient response of the LES structure to the ingition of the kicker engine will approximate a 10 g, 50 millisecond, inverse cosine shock pulse.	←115 sec.→										

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Table 4-3. Flight Mission Phases, Temperature Environmental Criteria

Spacecraft Section	Condition	Mission Phase: Lift Off and Boost	Earth Orbit	Translunar	Lunar Orbit	Transearth	Re-Entry
Command Module Interior (Cabin Wall)	Nominal	75 ± 5°F	75 ± 5°F	75 ± 5°F	75 ± 5°F	75 ± 5°F	140°F (See Figure II-A)
	Minimum	75 ± 5°F	0°F* (See Figure II-B)	0°F* (See Figure II-B)	0°F* (See Figure II-B)	0°F (See Figure II-B)	0°F
	Maximum	150°F	150°F* (See Figure II-C)	150°F* (See Figure II-C)	150°F* (See Figure II-C)	150°F* (See Figure II-C)	200°F
Parachute Compartment (Outer Wall - Region of Parachute with Parachute Installed)	Nominal	75 ± 5°F	40 to 60°F	40 to 60°F	40 to 60°F	40 to 60°F	150 to 200°F
	Minimum	60°F	-65°F	-65°F	-65°F	-65°F	0°F
	Maximum	175°F	120°F	120°F	120°F	120°F	250°F
Radome (Inner Wall)	Nominal	← DESIGN IN PROCESS →					
	Minimum	30°F	← DESIGN IN PROCESS →				
	Maximum	← DESIGN IN PROCESS →					151°F Fused Silica 180° Duriod (See Figure II-D)
Command Module Exterior (Ablation Material - Outer Honeycomb Interface)	Nominal	60 to 200°F	150 to 225°F (See Figure II-E)	-200 to 225°F	-150 to 225°F (See Figure II-F)	-200 to 225°F	500°F (See Figure II-A)
	Minimum	30°F	-200°F	-290°F (See Figure II-B)	-200°F	-290°F (See Figure II-B)	-290°F
	Maximum	270°F	250°F	250°F (See Figure II-C)	250°F	250°F (See Figure II-C)	600°F (See Figure II-A)
Service Module Interior**	Nominal	60 to 80°F	40 to 80°F	40 to 80°F	40 to 80°F	40 to 80°F	-
	Minimum	40°F	30°F (See Figure II-G)	30°F	30°F (See Figure II-G)	40°F	-
	Maximum	100°F	130°F***	130°F***	130°F***	130°F***	-
Service Module Exterior	Nominal	60 to 320°F (See Figure II-H	-100 to 150°F (See Figure II-I)	-150 to 200°F	-100 to 150°F (See Figure II-I)	-150 to 200°F	-
	Minimum	30°F	-150°F	-250°F	-150°F	-250°F	-
	Maximum	390°F (See Figure II-H)	200°F	250°F	200°F	250°F	-

*Under normal spacecraft operation, random orientation is assumed. However, for the emergency depressurized conditions it is assumed that an orientation will be imposed on the Command Module to maintain the cabin wall within the range 0° to 150°F.

**Assumes insulation is placed on inside surface of Service Module shell and radial shear panels.

***In vicinity of Service Module radiators

[redacted]

2. Radiation. This environment is not to be considered as design criteria.

3. Winds - Launching. The wind criteria for launching of the Saturn and Nova-type vehicles consist of a steady-state and a peak wind. The wind gradients below provides a 95-percent and a 99-percent launch capability during the strongest wind month for C-1 and C-5 vehicles.

Height	Steady-State Wind Knots C-1 (95 percent)	Steady-State Wind Knots C-5 (99 percent)	* Peak Winds Knots C-1 (95 percent)	* Peak Winds Knots C-5 (99 percent)
10	14.0	18.4	19.6	25.8
30	17.5	22.9	24.5	32.1
60	21.0	26.4	29.4	36.9
100	22.5	29.3	31.5	41.0
200	25.9	33.6	36.3	47.0
300	28.0	36.5	39.2	51.1
400	29.4	38.7	41.2	54.2

*Gust Characteristics. For the effects of gusts, a linear build up from the steady-state winds to the peak winds will be assured. The period of this build up and decay will be taken as 4 seconds for all height levels, i.e., build up of 2 seconds and 2 seconds for decay to steady-state winds speed.

4. Winds - Gusts Aloft. The in-flight wind criteria for the Saturn vehicles will be employed for the lunar occupancy payload studies. These criteria consist of a wind speed profile envelope, a wind shear spectrum envelope, and an allowance for gust and/or embedded jet characteristics.

   The wind speed profile envelopes for the 95-percent and a 99-percent probability level are illustrated in Figure 4-4 and the wind shears to be associated with these wind speeds are illustrated in Figure 4-5. The 95-percent wind speed and the 99-percent wind shears are the current (lunar orbit) criteria for the Saturn C-5 vehicle. Figures 4-6 and 4-7 illustrate how the above wind speed and wind shear data are combined to form a design synthetic wind profile. Winds are to be from the most adverse direction.

[redacted]

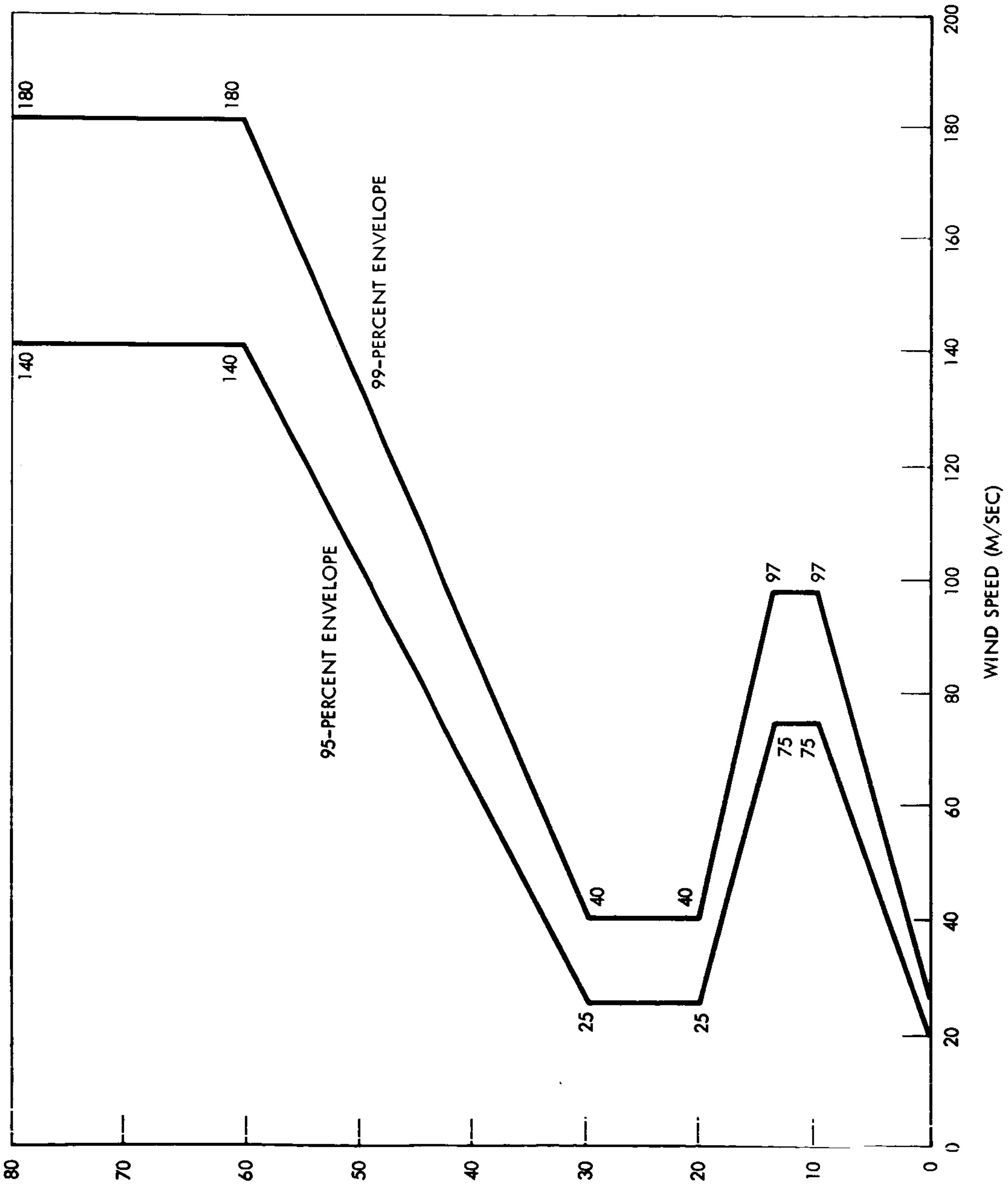

Figure 4-4. 95- and 99-Percent Wind Speed Envelopes

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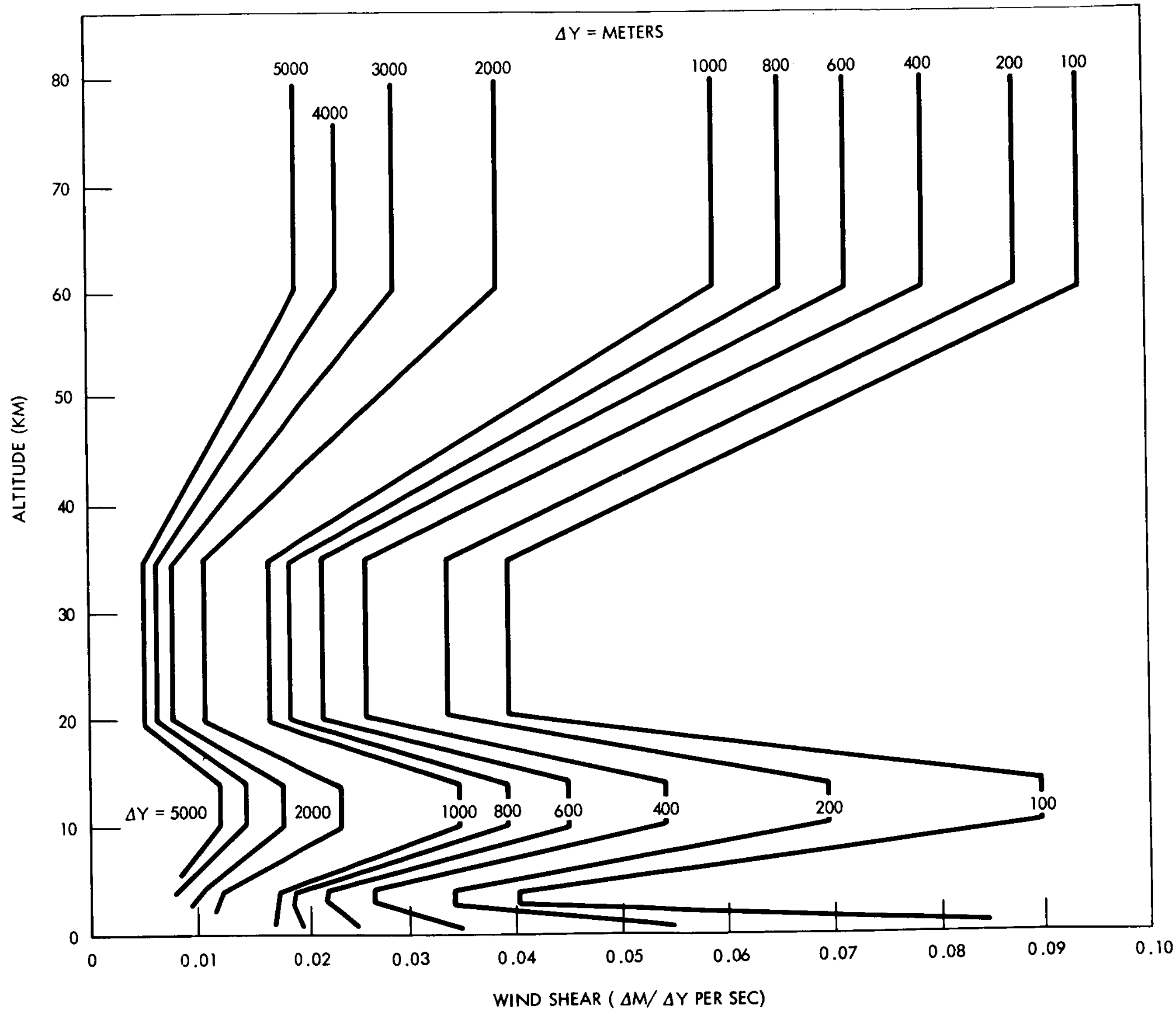

Figure 4-5. Ninety Nine-Percent Wind Shear Spectrum

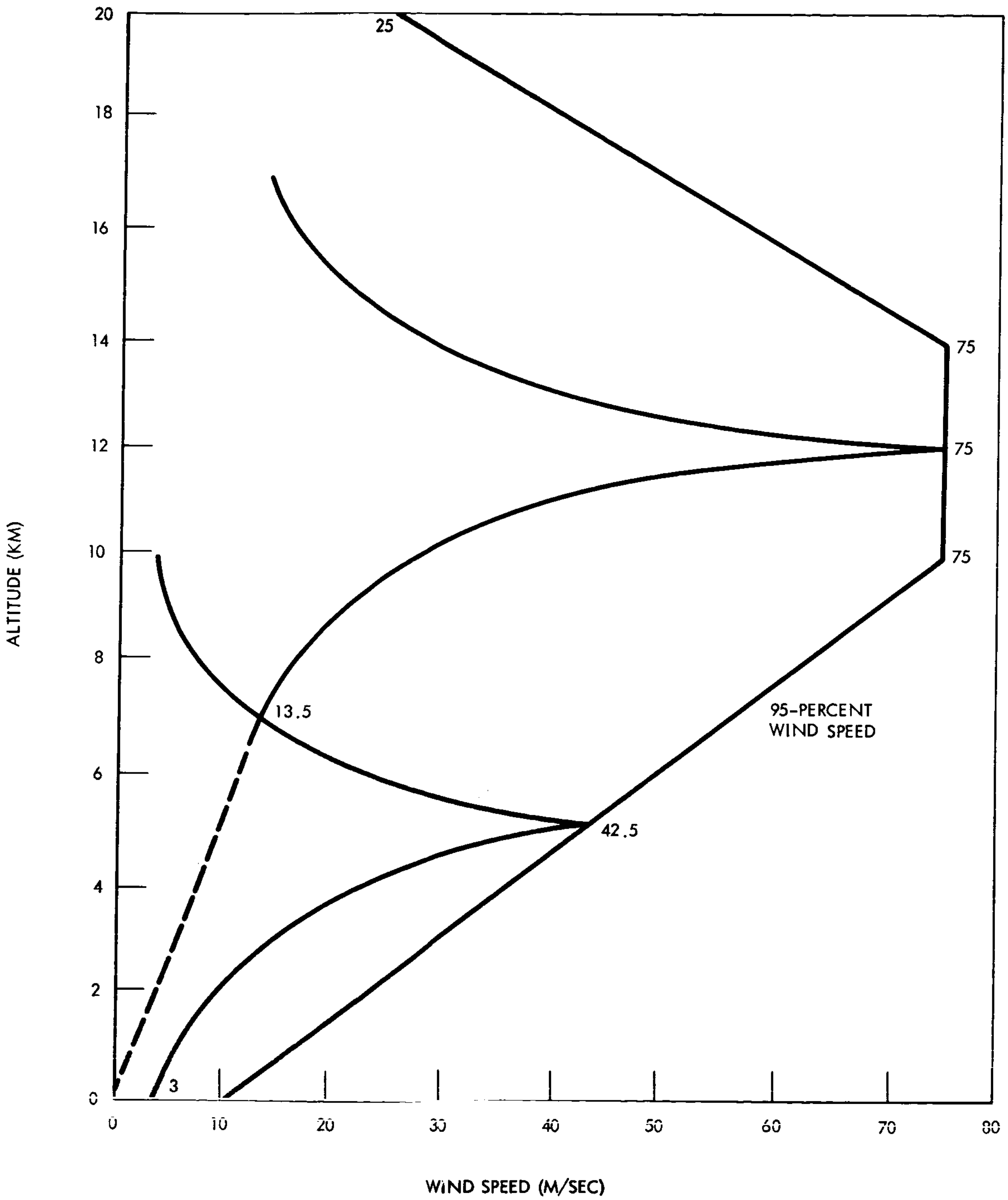

Figure 4-6. Ninety Five-Percent Wind Speed, 99-Percent Wind Shear Profiles (Altitudes of Concern, 5 and 12 KM)

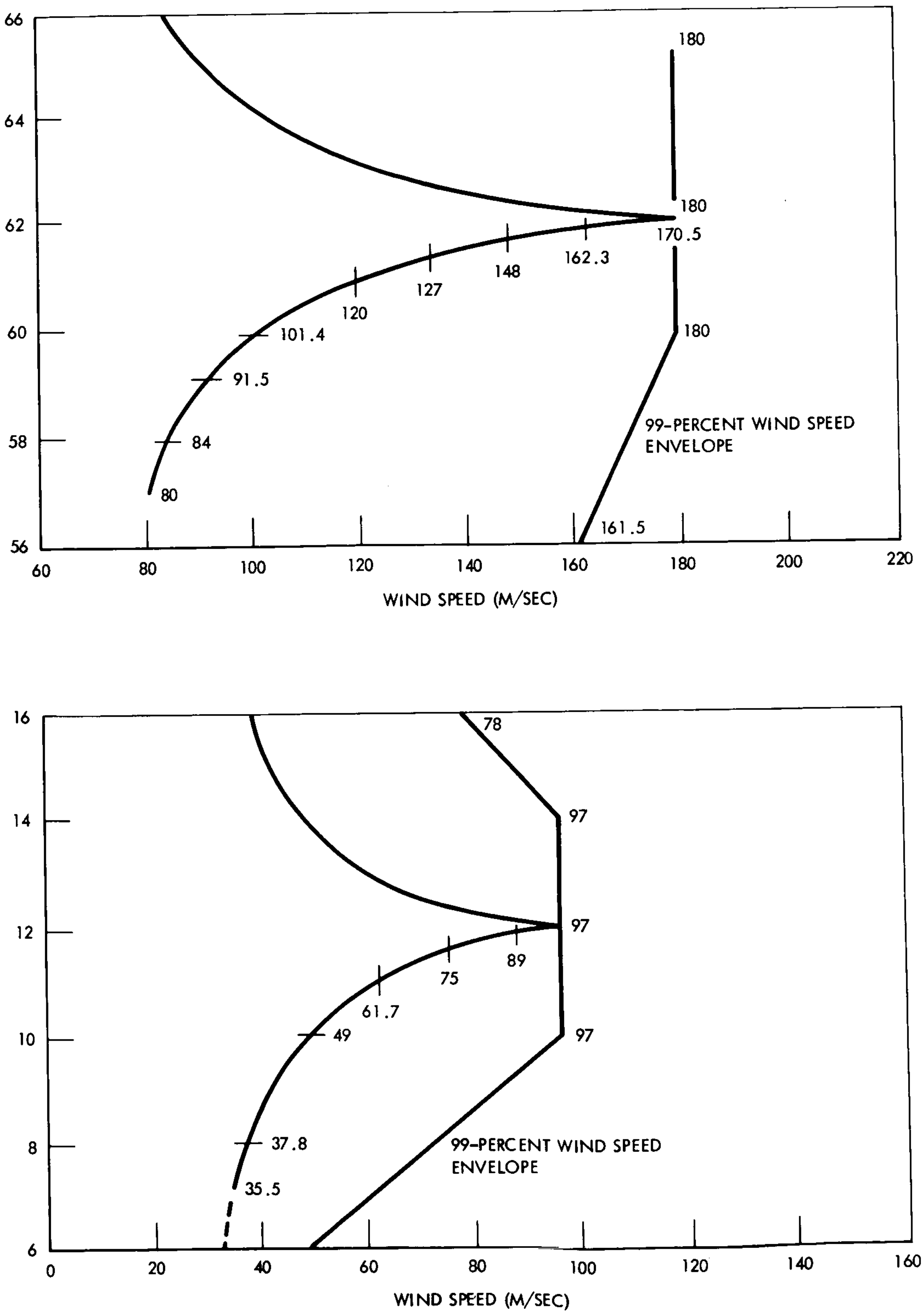

Figure 4-7. Ninety Nine-Percent Wind Speed, 99-Percent Wind Shear Profiles (Altitudes of Concern, 12 and 62 KM)

[redacted] [redacted]

An allowance must be made in design (structural and/or control) for gust and/or embedded jet characteristics in the 4-to-15-kilometer high dynamic pressure interval. A discrete or multiple (two or three) quasi-square wave shape wind increase with an amplitude of 9 meters per second should be employed. (See Figures 4-8.) This gust is to be applied so that the speed within the gust will reach a peak wind profile speed which is 9 meters per second greater than that of the synthetic wind profile. Wave length should be chosen according to the characteristics (structural and/or control) of the vehicle system. The discrete gust will be assumed to act normal to the space vehicle.

5. Humidity. A relative humidity of 30 percent to 70 percent is applicable to C/M interior under normal conditions (shirt-sleeve environment). A relative humidity up to 100 percent is applicable to S/M interior only during launch phase.

6. Meteoroids. This environment is not to be considered as design criteria.

7. Radio Interference (EMI). The requirements for interference control in the basic design of all electronic and electrical components, equipment, assemblies and systems will be in accordance with Specification MC 999-0002B.

8. Ocean Waves. This environment is applicable only to sea landing of the command module.

    a. Sea Conditions

    Low seas --------------1-to-3-foot waves

    Medium seas-----------3-to-8-foot waves

    High seas--------------8-foot or greater waves

    b. Swell Conditions

    Low swell -------------1 to 6 feet high

    Medium swell----------6 to 12 feet high

    High swell-------------12 feet or higher

[redacted]

[redacted]

[redacted]

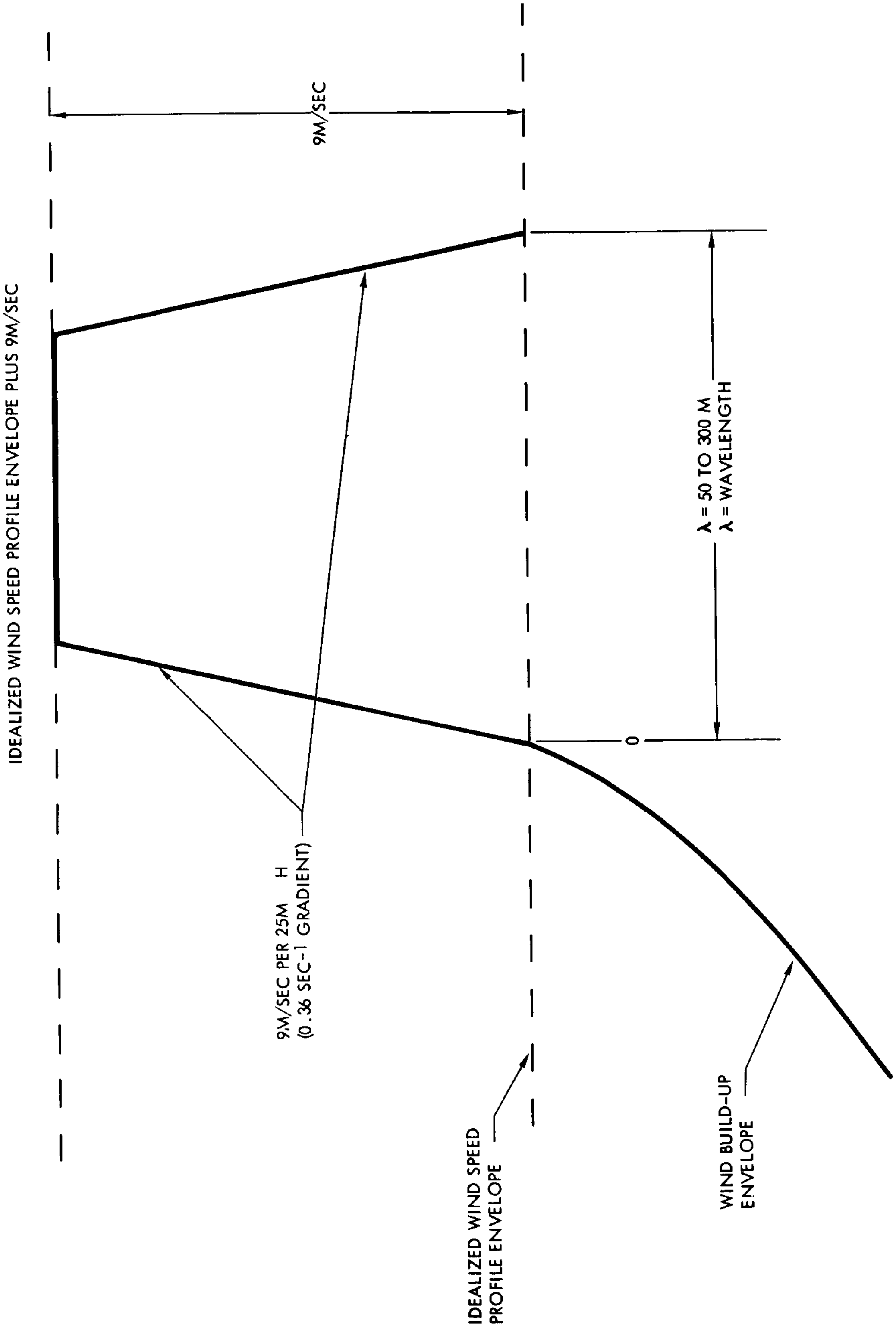

Figure 4-8. Relationship Between Established Gust Characteristics and the Idealized Wind Speed Profile Envelope

[redacted] [redacted]

9. Pressure

a. Command Module - Crew Compartment Only

Emergency conditions: 1×10^{-4} mm Hg for 100 hours

Normal conditions: 5 psi with 100 percent oxygen

b. Service Module

7.5×10^{-10} mm Hg for 360 hours

10. Pressure Differential

Command Module

Pressure Shell Burst: 9 psig
Crush: 8.3 psig

Fixed and Deployable H. S.
Burst: 3 psig
Crush: 8.3 psig

11. Hazardous Gases - Explosion Proofing. Design of equipment will be in accordance with the requirements of MSFC 10M01071.

The vibration criteria described in Table 4-2 have been revised. Note that certain criteria cover both the command module (less crew compartment) and the portion of the service module forward of station Xa equals 910. The criteria for the portion aft of station Xa equals 910 of the service module to station Xa equals 745 of the adapter are defined.

Included in this revision are criteria for the adapter and for the crew compartment.

4.2.4 Ground Support Equipment Environmental Levels

4.2.4.1 Operational Areas Environmental Criteria

The following environments will be experienced by GSE in the operational areas located in continental United States. These criteria do not apply to GOSS equipment.

[redacted]

[redacted]

[redacted]

The equipment will be designed or protection provided to prevent damage or operational degradation during and/or after exposure to these environments for 3 years. This criterion supplements that defined in paragraph 4.2.1.

Operational areas are defined as follows:

1. Unsheltered Area. An area exposed to the elements with no form of protection, such as preparation areas, launch areas, engine run-up areas, and storage areas

2. Sheltered Area. An environmentally controlled facility center, remote station, laboratory, etc.

3. Sun-Shaded Area. An out-of-doors area protected from exposure to the direct overhead sunlight by some stable or portable cover, such as may be found in a vehicle test stand or launch site

4.2.4.2 Unsheltered Area

1. Temperature. +15 to +105 F, plus solar radiation of 360 Btu/ft^2/hr for 6 hours/day for 2 weeks

2. Altitude. 6000 feet maximum

3. Relative Humidity. Up to 100 percent

4. Fungus. As experienced in tropical climates. The use of non-nutrient materials is recommended wherever possible. Where materials which are nutrients for fungi are used, such materials shall be treated with an approved fungicidal agent. Fungus resistant materials are defined in Specification MIL-E-5400.

5. Sand and Dust. As encountered in desert and ocean beach areas, equivalent to 140 mesh silica flour with particle velocity up to 500 ft/minute

6. Salt Spray. Salt atmosphere as encountered in ocean coastal areas, the effect of which is simulated by a 20 percent salt solution for 50 hours.

7. Rain. Up to 0.6 inches per hour for 12 hours

8. Hazardous Gases. Explosion proofing requirements, as defined in MSFC drawing 10M01071

9. Ozone. Up to 3 years exposure to 0.05 parts per million concentration

4.2.4.3 Sheltered Area

1. Temperature. Controlled - +60 to +80 F.

2. Temperature. Controlled but with environmental equipment out of commission - +52 to +105 F for 1 hour maximum

3. Altitude. 6000 feet maximum

4. Relative Humidity. 60 (+0, -15) percent at 70±10 F

4.2.4.4 Sun-Shaded Area

1. Temperature. +15 to 105 F

2. Altitude. 6000 feet maximum

3. Relative Humidity. Up to 100 percent

4. Sand and Dust. As encountered in desert and ocean beach areas, equivalent to 140 mesh silica flour with particle velocity up to 500 feet per minute

5. Fungus. As experienced in tropical climates. The use of non-nutrient materials is recommended wherever possible. Where materials which are nutrients for fungi are used, such materials shall be treated with an approved fungicidal agent. Fungus resistant materials are defined in Specification MIL-E-5400.

6. Rain. Up to 0.6 inch per hour for 12 hours

7. Salt Spray. Salt atmosphere as encountered in ocean coastal areas, the effect of which is simulated by a 20 percent salt solution for 50 hours

8. Hazardous Gases. Explosion proofing requirements, as defined in MSFC drawing 10M01071

9. Ozone. Up to 3 years exposure to 0.05 parts per million concentration

[redacted]

4.3 ENVIRONMENTAL TEST PROCEDURES

4.3.1 Application of Environments in Qualification Testing

This section describes a typical application of the environmental qualification test criteria. The program delineated herein forms the basis of all the test programs as applied to the various subsystems and components. Specific application to any given subsystem is delineated under the pertinent paragraph of Section 5. A complete qualification test summary is shown by test phase in Table 4-4. The table also shows the manner in which each test component is scheduled and controlled by sample or serial number.

4.3.2 Acceleration

4.3.2.1 General

1. Levels - design proof
    a. Equipment (S/M) - 7 g
    b. Equipment subjected to reentry (C/M) - 20 g applied only to those which must operate during or after abort and 7 g applied to all other C/M equipment
2. Time - 5 minutes in both directions in each of the three mutually perpendicular axes
3. Levels - off-limit
    a. Equipment (S/M) - 10 g
    b. Equipment subjected to reentry (C/M) - 26 g applied only to those which must operate after abort and 10 g applied to all others

4.3.2.2 Detail

The test specimen will be mounted on the centrifuge in the position specified in the applicable equipment specification. If the starting position is not specified, the bottom and front of the test unit will be arbitrarily assigned, and the specimen will be mounted upright facing in the direction of angular motion. The centrifuge will be brought up to the rotational speed required to produce the radial acceleration, and this acceleration will be stabilized and maintained for a period of not less than 5 minutes. The test specimen will then be rotated, so that it will be subjected to the

Table 4-4. Qualification Test Summary by Test Phase

Phase		Test Title	NOTE: (1) (2) (3) and (4) indicate S/C zones which apply.	Specimen Utilization Total Samples			
				4	6	8	10
Climatics and Checkout		Acceptance	(1) (2) (3) (4)				
		Humidity	(1) (2) (3) (4)				
		Sand and Dust	(Exterior of S/C only)				
		Salt Fog	(Exterior of S/C only)				
		Explosion	(1) (2) (3) (4)				
		Electromagnetic Interference	(1) (2) (3) (4)				
Design Proof	Mission Sequential Maximum Levels	Acceleration	(1) (2)	Sample No. 1	Sample No. 1 & 2	Sample No. 1, 2, 3	Sample No. 1, 2. 3
		Vibration	(1) (2) (3) (4)				
		Acoustics	(1) (2) (3) (4)				
		High Temperature	(1) (2) (3) (4)				
		Low Temperature	(1) (2) (3) (4)				
		Vacuum	(1) (2) (3) (4)				
		O_2 Atmosphere	(2)				
		Thermal Shock	(1) (2) (3) (4)				
		Shock	(2)				
Phase A	Off-Limit	Vibration	(1) (2) (3) (4)	Sample No. 2 & 3	Sample No. 3 & 4	Sample No. 4 & 5	Sample No. 4, 5, 6
		High Temperature	(1) (2) (3) (4)				
		Low Temperature	(1) (2) (3) (4)				
		Operation to Failure	(1) (2) (3) (4)				
		Shock	(2)				
Phase B - Life	Mission Sequential Normal Levels	Acceptance	(1) (2) (3) (4)	Sample No. 3 & 4	Sample No. 5 & 6	Sample No. 6, 7, 8	Sample No. 7, 8, 9 & 10
		Checkout Life	(1) (2) (3) (4)				
		Acceleration	(1) (2) (3) (4)				
		Vibration	(1) (2) (3) (4)				
		Acoustics	(1) (2) (3) (4)				
		Vacuum High Temp.	(1) (2) (3) (4)				
		Vacuum Low Temp.	(1) (2) (3) (4)				
		O_2 Atmosphere	(2)				
		Thermal Shock	(1) (2) (3) (4)				
		Shock	(2)				

required acceleration in each direction along each of the mutually perpendicular axes. If any special axis is considered significant with respect to acceleration effects, the axis will be defined and the acceleration specified applied to the geometric center of the specimen. The centrifuge arm (measured to the geometric center of the specimen) will be at least five times the dimension of the specimen (measured along the arm).

4.3.3 Shock

4.3.3.1 General (C/M only)

1. Levels

   a. Design proof 78 g

   b. *Mission life ____g (Where applied, the test level will be determined by S&ID as a function of the component's specific location.)

2. Waveform. Sawtooth with linear rise time of 11 ± 1 milliseconds and decay time 1 ± 1 milliseconds

3. Application

   a. Design proof. One direction in each of the three mutually perpendicular axis

   b. *Mission life. Once in each of the two major axes (longitudinal and radial) anticipated during impact

   *This test is applied only to equipment that is required to function after landing impact.

4.3.3.2 Detail. The test fixture is to be designed such that any resonances inherent in the fixture are minor. The test specimen will be attached to a fixture capable of transmitting the shock conditions specified to the equipment under test. The attachment to the test fixture will be the space vehicle mounting or simulate it, where necessary. The shock pulse will be monitored on the test fixture near the specimen mounting points. During the shock tests, the shock waveform timing pulses, shock amplitude calibration line, and, if required, the operation of test specimen shall be monitored and recorded either on a photograph or a recording oscillograph. The shock wave, the timing pulses, and the operation of the test specimen shall be recorded simultaneously. There shall be a complete record of each shock pulse.

[redacted]

4.3.4 Oxidation

4.3.4.1 General

1. Pressure. For design proof equals 1 atmosphere and for mission simulation life equals 5 psia

2. Time. For design proof equals 4 hours and for mission simulation life equals one mission duty cycle

3. Gas. 100 percent O_2

4.3.4.2 Detail

1. Preparation. Due to the possible hazard, the test equipment will be an explosion test chamber built to MIL-E-5272C requirements. The equipment to be tested will be installed in the test chamber in such a manner that normal electrical operation is possible and mechanical controls may be operated through the pressure seals from the exterior of the chamber. All external covers of the test specimen will be removed or opened to insure circulation of the O_2. The equipment shall be operated to determine that it is functioning properly and to energize potential sparking or high temperature components. Mechanical loads on drive assemblies and servos, mechanical and electrical loads on switches, and relays may be simulated, when necessary, if the proper precaution is taken to duplicate the normal loads with respect to torque, voltage, current, and inductive reactance. In all instances, it will be considered preferable to operate the equipment as it normally functions in the system during service use.

2. Test Procedure. The test chamber will be sealed and the internal pressure of 1 atmosphere or 5 psia will be maintained with 100 percent oxygen. In the case of the mission simulation-life test, the equipment will be operated in a manner which simulates operation during a mission. The test specimen will then be operated with the making and breaking of all electrical contacts. If high temperature components are present, sufficient warm-up time will be employed. The test time will be determined and specified by S&ID.

[redacted]

[redacted]

4.3.5 Vacuum

4.3.5.1 General

1. Pressure. Design proof - 1×10^{-4} mm Hg (C/M interior)
1×10^{-6} mm Hg (S/M interior)
7.5×10^{-10} mm Hg * (external to S/C)

2. Time. Design proof equals 100 hours

*Hard Vacuum test applies only to small vacuum sensitive components and to materials. Larger equipment, subsystems, black boxes, etc., shall be subjected to vacuum of 1×10^{-6} mm Hg.

4.3.5.2 Detail

The equipment shall be placed in the test chamber and the pressure reduced to the selected level. This pressure will be maintained as determined by S&ID. If the equipment is to operate in service under vacuum conditions, it shall be operated in a manner simulating lunar mission performance requirements. (The allowable performance variation will be defined in the applicable procurement or process specification.) Pressure will then be raised to ambient level. The equipment will be visually inspected for deterioration and operated to determine compliance with the performance requirements of the pertinent specification.

4.3.6 Temperature (Low or High)

4.3.6.1 General

The test level and time will depend on the location of the test article in the vehicle and the mission time during which it must operate.

4.3.6.2 Detail

The equipment will be placed within the test chamber. The internal temperature of the chamber will be brought to the selected temperature. The equipment will be exposed to that temperature for the specified time. Equipment operation will simulate the in-service requirements. Performance during and/or after exposure to this test must conform to the requirements of the applicable equipment specification. The equipment will then be returned to ambient conditions and the equipment again operated and inspected visually. (If there is a specific rate of temperature change, this will be included in the individual test procedure.)

4.3.6.3 Cold Plates

For equipment that is temperature controlled by cold plates, the temperature tests are conducted as described below:

1. Design Proof. Equipment that must operate during the lunar mission, with the exception of the reentry and recovery phase will be inserted in the temperature chamber with cold plates attached and stabilized at the selected ambient air temperature with a coolant inlet temperature and a rate of coolant flow as determined by S&ID for the specific component. (Note: This will be the maximum inlet temperature anticipated.) The operation of the equipment and the outlet coolant and internal temperature will be monitored during this test. The procedure described above shall then be repeated with an ambient air temperature and a coolant inlet temperature, which will represent the minimum temperature anticipated. The flow rate shall remain constant throughout the tests.

2. Recovery Essential Equipment. Equipment that must operate during reentry and recovery will be stabilized to selected ambient air and inlet coolant temperatures and coolant flow rate. The flow of coolant will be terminated and the ambient air temperature will be increased linearly to a predetermined temperature in a specified time which would simulate the ambient rise experienced during reentry. These temperatures will be maintained for a predetermined length of time. This will be followed by a decrease in temperature for a specified time commensurate with landing and recovery phases.

3. Off-Limit. The equipment will be stabilized to selected ambient air and inlet coolant temperatures and coolant flow rate. The equipment will then be operated and allowed to stabilize at its new temperature. The equipment will be monitored for proper operation. The coolant temperature will be increased by a predetermined amount and the equipment temperature allowed to stabilize. The ambient air temperature then will be increased by a predetermined amount and the temperature of the equipment allowed to stabilize again. The steps in this paragraph will be repeated in the same sequence with the coolant temperature, and the ambient air temperature increased in predetermined increments.

[redacted]

4.3.7 Vibration

4.3.7.1 General

Level and Time

Design proof (See Tables 4-5 and 4-6)
Off-limit (See Table 4-7)
Mission simulation-life (see Table 4-8)

4.3.7.2 Detail

4.3.7.2.1 Mounting Description. The test specimen will be attached to a test fixture capable of transmitting the vibration conditions specified herein. Attachment of the test specimen to the test fixture will duplicate the space vehicle mounting or simulate it, where necessary. The test fixture will be designed such that any inherent resonances within the test frequency band will be minor. Any cables, tubing, or wires connected to the assembly under test will not affect the vibration inputs. The magnitude of the applied vibration will be monitored on the test fixture near the specimen mounting points. Vibration accelerometers will be mounted triaxially. Equipment response will be monitored in three major axes and other critical areas, as required.

4.3.7.2.2 Resonant Frequencies. Resonant modes of the assembly under test will be determined by varying the frequency of applied vibration slowly through the frequency range 5 to 2000 cps at a minimum amplitude (2 g or less).

Precautions will be taken to avoid exceeding the specified vibration input at unexpected resonant frequencies. A resonant search will be made in each of the three mutually perpendicular axes. Any resonant frequency that is accompanied with a gain of 1.5 or greater will be recorded and noted in the test report.

4.3.7.2.3 Combined Random and Sinusoidal Vibration. The test specimen will be subjected to a combined vibration test consisting of a sinusoidal waveform superimposed on a random waveform. The vibration will be applied successively in each of the three mutually perpendicular axes. The vibration levels and times are specified in Tables 4-5, 4-6, 4-7 and 4-8 and are graphically displayed in frequency and time domains in Figures 4-9 and 4-10. The sinusoidal sweep will be from 5 to 2000 cps and will be applied at a logarithmic rate. At the conclusion of the test, the test specimen will be inspected for any visible internal or external damage which could prevent the test item from meeting its operational reliability requirements. If a malfunction occurs during or after the environmental test, an internal inspection will be conducted to provide the required failure analysis data.

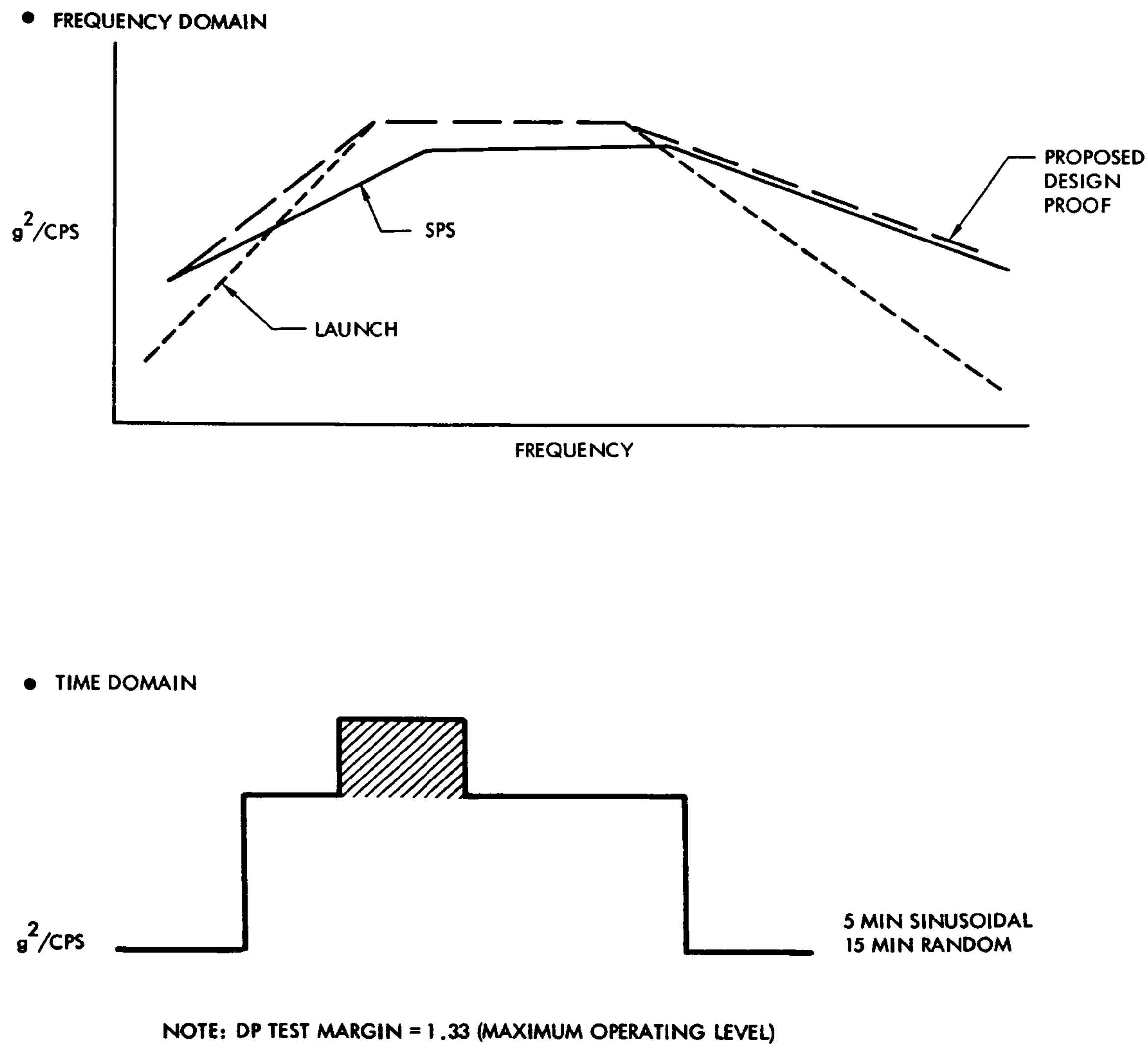

Figure 4-9. Design Proof Level

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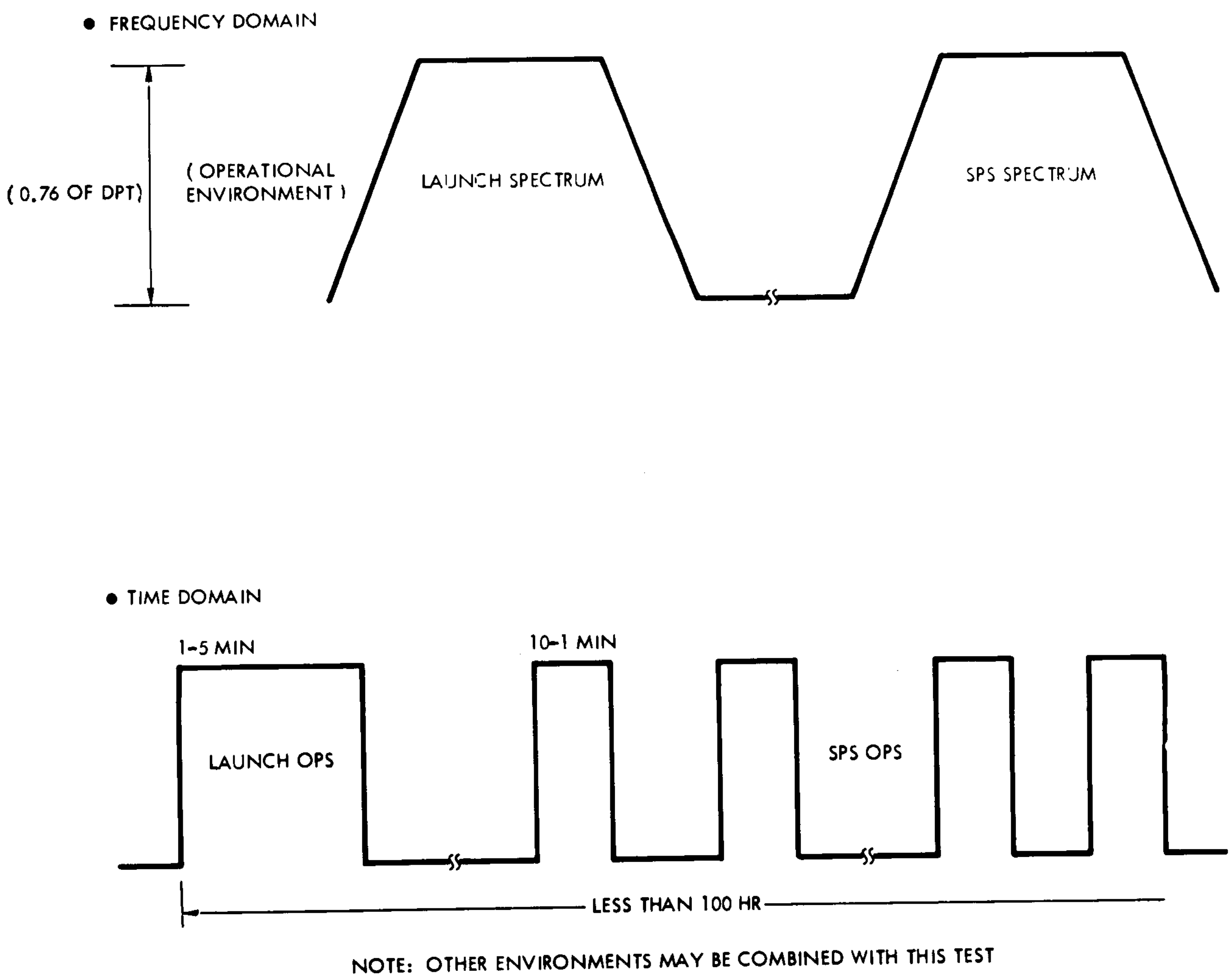

Figure 4-10. Mission Simulation Life Application

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Table 4-5. Vibration Test Criteria for Design Proof Tests

Vibration	Zone 1	Zone 2	Zone 3	Zone 4
Random (linear relates) to plot on log-log seals	5 to 50 cps linear increase from 0.007 to 0.122 g^2/cps 50 to 150 cps, constant at 0.122 g^2/cps 150 to 2000 cps, linear decrease from 0.122 to to 0.035 g^2/cps	5 to 50 cps, linear increase from 0.007 to 0.095 g^2/cps 50 to 150 cps, constant at 0.095 g^2/cps 150 to 2000 cps, linear decrease from 0.095 g^2/cps to 0.015 g^2/cps	5 to 50 cps, linear increase from 0.009 to 0.086 g^2/cps 50 to 150 cps, constant at 0.086 g^2/cps 150 to 2000 cps, linear decrease from 0.086 g^2/cps to 0.058 g^2/cps	5 to 50 cps, linear increase from 0.004 to 0.025 g^2/cps 50 to 150 cps, constant at 0.025 g^2/cps 150 to 2000 cps, linear decrease from 0.025 g^2/cps to 0.005 g^2/cps
Sinusoidal (g Peak) (linear relates to plot on log-log scale)	5 to 70 cps, linear increase from 0.3 g to 6 g 70 cps to 2000 cps constant at 6 g	5 to 70 cps, linear increase from 0.3 g to 5 g 70 to 1000 cps, constant at 5 g 1000 cps to 2000 cps constant at 4 g	5 to 90 cps, linear increase from 0.25 g to 5 g 90 to 2000 cps, constant at 5 g	5 to 100 cps, linear increase from 0.22 g to 3 g 100 to 600 cps, constant at 3 g 600 to 2000 cps constant at 4 g

NOTE: (1) Test time 15 minutes/axis, random 5 minutes/axis sinusoidal.

(2) Vibration levels shown are for basic structure and for small rigidly mounted packages. For rigidly mounted packages with weights in excess of 10 pounds, correct vibration levels must be specifically established.

(3) Zone 1 - Forward and Aft C/M equipment compartments to Xa 910 on S/M
Zone 2 - Crew Compartment Interior
Zone 3 - S/M and Adapter from Xa 910 to Xa 745
Zone 4 - Adapter - Aft of Xa 745

[redacted]

Table 4-6. Vibration Test Criteria for Design Proof Tests, Max q Abort

Vibration	Zone 1	Zone 2
Random (linear relates to plot on log-log scale)	5 to 60 cps, linear increase from 0.006 to 0.155 g^2/cps	5 to 60 cps, linear increase from 0.006 g^2/cps to 0.13 g^2/cps
	60 to 500 cps, constant at 0.155 g^2/cps	60 to 200 cps, constant at 0.13 g^2/cps
	500 to 2000 cps, linear decrease from 0.155 to 0.35 g^2/cps	200 to 2000 cps, linear decrease from 0.13 g^2/cps to 0.006 g^2/cps
Sinusoidal (g peak) (linear relates to plot on log-log scale)	5 to 90 cps, linear increase 0.3 to 8 g	5 to 90 cps, linear increase 0.3 to 7 g
	90 to 300 cps, constant 8 g	90 to 2000 cps, constant 7 g
	300 to 2000 cps, constant 12 g	

Note: Equipment that must operate during or after a Max-q abort and the failure of which will affect crew safety will be subjected to the vibration test described for a period of 5 minutes in the equipment's most sensitive axis (determined during development). If the most sensitive axis is not determined in the development test, vibrate the test specimen in the axis of vehicle thrust.

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Table 4-7. Vibration Test Criteria for Off-Limit Tests

Level	Criteria
Level 1	The random input spectrum of the design proof test (See Table 4-2) with (1) the random rms g level increased by 50 percent and (2) the sinusoidal input omitted
Level 2	The level shall be 175 percent of the random design proof rms g level.
Level 3	The level shall be 200 percent of the design proof level.
Level 4	The level shall be 225 percent of the design proof level.
Level 5	The level shall be 250 percent of the design proof level.
	Test time is 5 minutes per test level in the axis of vehicle thrust.

[redacted]

[redacted]

Table 4-8. Vibration Test Criteria for Mission Simulation-Life Tests

PART 1 Vibration	Zone 1	Zone 2	Zone 3
Random (linear relates to log-log plot)	5 to 50 cps linear increase from 0.008 to 0.93 g^2/cps 50 to 150 cps constant 0.093 g^2/cps 150 to 2000 cps linear decrease from 0.093 to 0.005 g^2/cps	5 to 50 cps linear increase from 0.007 to 0.072 g^2/cps 50 to 150 cps constant 0.072 g^2/cps 150 to 2000 cps linear decrease from0.072to 0.003 g^2/cps	5 to 50 cps linear increase from 0.009 to 0.065 g^2/cps 50 to 150 cps constant 0.065 g^2/cps 150 to 2000 cps linear decrease from 0.065 to 0.004 g^2/cps
PART 2 Vibration	**Zone 1**	**Zone 2**	**Zone 3**
Random (linear relates to log-log plot)	5 to 100 cps linear increase from 0.007 g^2/cps to 0.04 g^2/cps 100 to 550 cps constant 0.04 g^2/cps 550 to 2000 cps linear decrease to 0.027 g^2/cps	5 to 100 cps linear increase from 0.007 g^2/cps to 0.031 g^2/cps 100 to 200 cps constant 0.031 g^2/cps 200 to 2000 cps linear decrease to 0.011 g^2/cps	5 to 100 cps linear increase from 0.009 g^2/cps to 0.053 g^2/cps 100 to 550 constant at 0.053 g^2/cps 550 to 2000 cps linear decrease to 0.038 g^2/cps

NOTE: (1) Test time for Part 1 shall be 5 minutes

(2) Test time for Part 2 shall consist of ten (10) one (1) minute bursts with adequate time between each burst for inspection and/or monitoring of test specimen.

(3) Test levels of packages in excess of 10 pounds must be specifically established.

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Table 4-9. Acoustic Test Criteria for Design Proof and Mission Simulation-Life Tests

Frequency (cps)	Location and Levels (db)				
	Zone 1 External (vehicle)	Zone 3	Zone 1 (internal)	Zone 2	Zone 4
37.5 - 75	161 (+3, -0)	143 (+3, -0)	150 (+3, -0)	133 (+3, -0)	141 (+3, -0)
75 - 150	158 (+3, -0)	140 (+3, -0)	147 (+3, -0)	131 (+3, -0)	138 (+3, -0)
150 - 300	155 (+3, -0)	136 (+3, -0)	145 (+3, -0)	127 (+3, -0)	134 (+3, -0)
300 - 600	150 (+3, -0)	132 (+3, -0)	140 (+3, -0)	122 (+3, -0)	130 (+3, -0)
600 - 1,200	145 (+3, -0)	122 (+3, -0)	132 (+3, -0)	116 (+3, -0)	123 (+3, -0)
1,200 - 2,400	139 (+3, -0)	119 (+3, -0)	125 (+3, -0)	109 (+3, -0)	116 (+3, -0)
2,400 - 4,800	133 (+3, -0)	113 (+3, -0)	120 (+3, -0)	101 (+3, -0)	110 (+3, -0)
4,800 - 9,600	129 (+3, -0)	107 (+3, -0)	116 (+3, -0)	95 (+3, -0)	104 (+3, -0)
Over-all	168.5 (-3, +0)	148 (-3, +0)	158 (-3, +0)	140 (-3, +0)	147 (-3, +0)

Note: The levels shown will be the test levels for design proof tests A-1 and for life phase B. The test may be omitted from phase B if there will be no fatigue as a result of acoustics.

Time for test is 5 minutes.

[redacted]

Table 4-10. Acoustic Test Criteria for Abort-Essential Equipment

Frequencies	Location and Levels (db)		
	Zone 1	Zone 2	External
37.5 - 75	150 (+3, -0)	134 (+3, -0)	161 (+3, -0)
75 - 150	150 (+3, -0)	134 (+3, -0)	161 (+3, -0)
150 - 300	149 (+3, -0)	131 (+3, -0)	159 (+3, -0)
300 - 600	144 (+3, -0)	129 (+3, -0)	157 (+3, -0)
600 - 1,200	145 (+3, -0)	130 (+3, -0)	158 (+3, -0)
1,200 - 2,400	147 (+3, -0)	129 (+3, -0)	159 (+3, -0)
2,400 - 4,800	144 (+3, -0)	126 (+3, -0)	157 (+3, -0)
4,800 - 9,600	142 (+3, -0)	122 (+3, -0)	155 (+3, -0)
Over-all	158 (-3, +0)	142 (-3, +0)	171 (-3, +0)

Note: This test is to be performed on equipment that must operate during or after an abort and the failure of which will affect crew safety.

Test time is 1 minute.

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4.3.8 Acoustics

4.3.8.1 General

The acoustic levels are delineated by zones in Tables 4-9 and 4-10.

4.3.8.2 Detail

4.3.8.2.1 Apparatus. The assembly will be placed within a reverberant test enclosure suitably formed and proportioned to produce, as closely as possible, a diffuse sound field, the sound energy density of which is very nearly uniform throughout the enclosure. Acute angles of adjacent walls will be avoided, whenever possible.

4.3.8.2.2 Mounting. The test specimen will be suspended in the test enclosure by means of a soft suspension device, such as soft springs or elastic cord. The natural frequency of all modes of suspension will be less than 25 cps. Every surface of the test specimen will be exposed to the sound field by centrally locating it in the test chamber. The test specimen volume will be no greater than 10 percent of the test chamber volume. If the test chamber is rectangular, no major surface of the test specimen will be installed parallel to a chamber wall.

4.3.8.2.3 Procedure. The sound pressure field will be measured with the test item mounted in the test chamber. Measurements will be made by using a microphone (or more than one as required) to measure the test specimen sound field in the proximity of each major dissimilar surface. The over-all sound pressure level will be within -3 db of the test level, and the time to conduct the survey or measurements will be less than 1/10 of the final test time. If possible, the test microphones will be moved over the test specimen surface at least 18 inches from the surface of the test article.

The over-all sound pressure level will be raised to the correct test level. The average sound pressure distribution around the test specimen will be uniform within 0 to +3 db of the desired value. The test time shall be as shown in Tables 4-9 or 4-10. If the unit under test is to operate while subjected to the acoustic environment, operating the test article will be monitored during this test. The test data will be compared to the preceding operational test at room ambient conditions. At the conclusion of the test, the test specimen will be inspected for any visible internal or external damage which could prevent the test item from meeting its operational and reliability requirements. It should be noted that an internal inspection may be impractical and may be omitted prior to operation of the unit. If a malfunction occurs during or after being subjected to the acoustic environment,

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internal inspection will be performed as failure analysis requirements dictate. If the test specimen malfunctioned during the test but operated within functional requirements after removal of the acoustic environment, a single frequency sweep sound test will be performed to determine how the malfunction can be duplicated and its specific cause. In the application of the single frequency sound, the sound pressure field shall be measured as specified above. A single frequency sound at which a similar malfunction is observed shall be recorded and the results compared with the data obtained during the continuous spectrum test.

4.3.9 Humidity (Application Limited)

4.3.9.1 General

1. Level. 95 percent (+5, -0) percent, relative humidity

2. Temperature. 68 - 160 F

3. Time. 240 hours

4.3.9.2 Detail

Procedure per MIL-Std 810

4.3.10 Electromagnetic Interference

4.3.10.1 General

As in S&ID specification MC 999-0002B

4.3.10.2 Detail

As in S&ID specification MC 999-0002B

4.3.11 Sand and Dust (Application Limited)

4.3.11.1 General

As encountered in beach and coastal areas, particularly AMR

4.3.11.2 Detail

Procedure per MIL-Std 810

[redacted]

[redacted]

4.3.12 Salt Fog (Application Limited)

4.3.12.1 General

As encountered in beach and coastal areas, particularly AMR

4.3.12.2 Detail

Procedure per MIL-Std 810

4.4 COMBINED ENVIRONMENTAL TESTS

The combined environmental tests are those tests used to simulate actual spaceflight conditions as nearly as possible. The procedures will be applied as facilities and schedules permit. Particular attention is to be given those components where the failure mode analysis indicates a possible problem area.

4.4.1 Temperature-Vacuum

When a combined temperature, vacuum, and a dynamic test is to be performed, the test specimen will be stabilized at the required temperature, and the required pressure obtained. Then the dynamic test will be performed. If the unit is to be operated during the combined test operation, it will first be checked after the temperature has stabilized, and the required pressure has been achieved.

4.4.2 Vibration-Acceleration

The test unit is to be mounted on a centrifuge-vibration test equipment. The test specimen is to be operated to determine compliance with its procurement specification. It will then be subjected to the acceleration environment. While undergoing this environment, it shall be operated if it will operate in service during acceleration. The vibration test will then be performed per paragraph 4.3.7. Note that equalization of the vibration equipment shall be accomplished while the equipment is subjected to the acceleration environment.

4.4.3 Temperature-Vibration

With the test unit mounted on the vibration test equipment and installed within the temperature chamber, the test specimen will be operated to determine compliance with the requirements of its procurement specification. It is then to be stabilized at the required test temperature. At this time the units operation must be checked again. After it has been determined that the temperature environment does not cause a failure, the combined temperature-vibration test will be performed, as required, for vibration alone in paragraph 4.3.7.

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5.0 SUBSYSTEM QUALIFICATION TESTS*

This section presents the individual sybsystem qualification test plans, and features the variant physical characteristics and unique mission requirements of each subsystem. Each test plan has been designed to verify system capability to perform within design specification requirements for a sufficient length of time to fulfill mission operational requirements. Another test plan objective is to demonstrate repeatability of functional and quality characteristics. Each test plan has been made optimum to qualify the respective equipment in ground simulation within the program constraints as defined in previous sections of this volume. Reliability assessment models, listings of system parameters for qualification testing, and qualification test schedules are delineated for each subsystem test plan. The qualification test schedules are defined within the limitations of information available at this issue of the Apollo General Test Plan. Schedules will be revised to reflect the current program direction and requirements.

5.1 SERVICE PROPULSION SYSTEM

5.1.1 Scope

The qualification test plan for the service propulsion system will include environmental tests on components and the complete engine. It will consist of a sea level phase and a simulated altitude phase. Each test will employ an engine equipped with a thrust chamber that had been fired for a duration which did not exceed 30 seconds (accumulated during acceptance testing). Additional tests will be conducted under abnormal conditions and environments for extended periods to exhibit equipment design margin capabilities. Selection of the number of engine tests is based on the requirement to assess engine capabilities under simulated mission environments and off-limit operating conditions.

5.1.2 Reliability Assessment Model

Figure 5-1 depicts the reliability assessment model for the service propulsion system. The model will aid in establishing test requirements and in programming the collection of data for the assessment of reliability by illustrating the interfaces of system components and subassemblies.

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5.1.3 Criticality and Hardware Requirements

5.1.3.1 Service Propulsion Engine

Table 5-1 designates the criticality class, as defined in subsection 2.4, for the components of the service propulsion engine. The table enumerates the hardware requirements for a minimum qualification program as established in subsection 2.5. The location of the components in the spacecraft is indicated by spacecraft zone numbers as defined in subsection 4.2.

Table 5-1. SPS Engine Parameters for Qualification Testing

Component	S/C Zone	Criticality	Required Hardware
Thrust chamber	3	1	24
Valves	3	1	13
Actuator	3	1	14
Mount	3	1	11
Lines	3	1	11
Nozzle	4	1	6
Injector	3	1	12

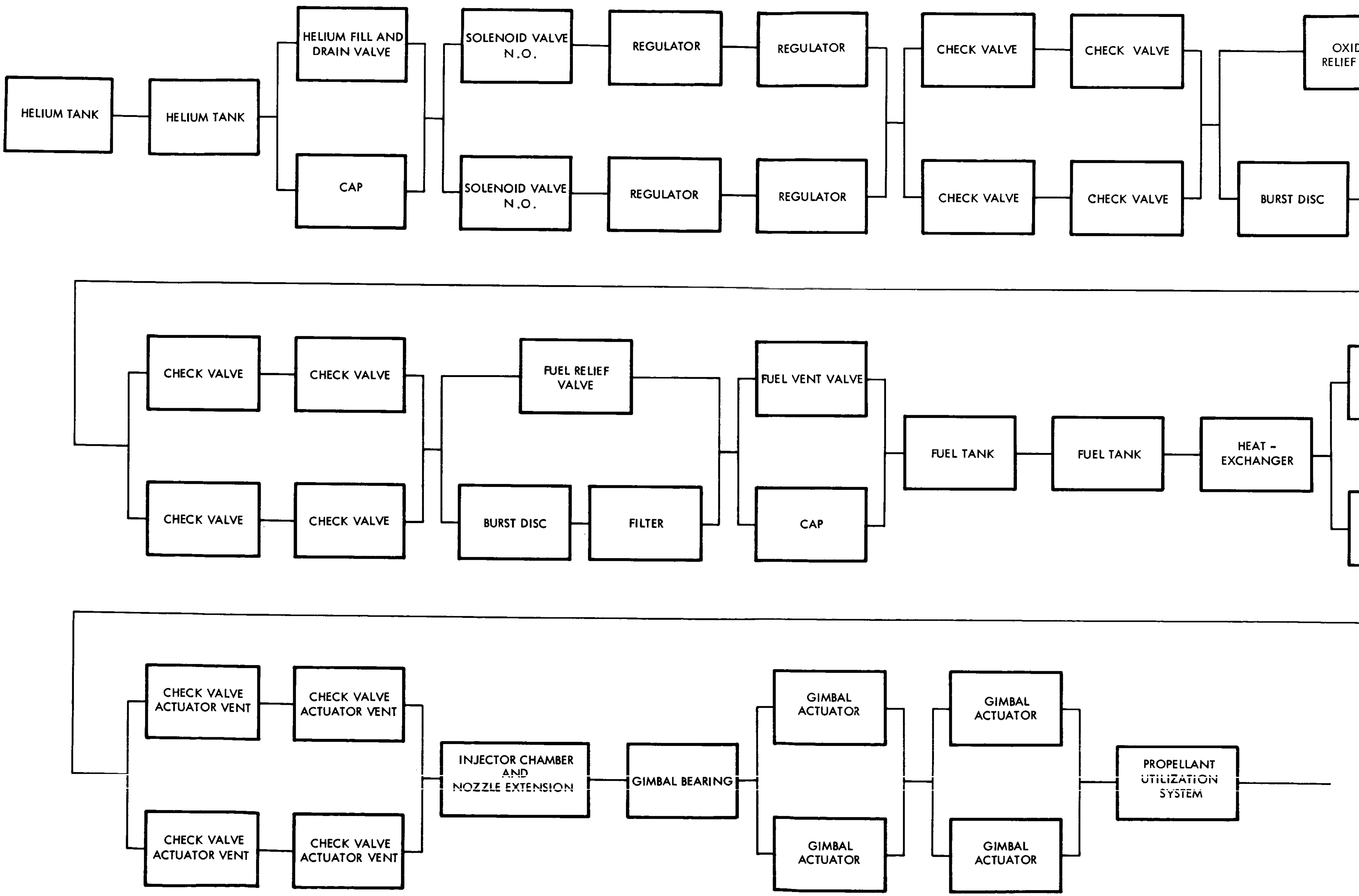
5.1.3.2 Propellant Feed

Table 5-2 designates the criticality class, as defined in subsection 2.4, for the components of the service propulsion system propellant feed. The table enumerates the hardware requirements for a minimum qualification program as established in subsection 2.5. The distribution of the hardware through the subsystem qualification test phases, as described in subsection 2.3, is shown. The location of the components in the spacecraft is indicated by spacecraft zone numbers as defined in subsection 4.2.

5.1.4 Qualification Test Schedule

5.1.4.1 Service Propulsion Engine

Figure 5-2 displays the qualification test schedule for the service propulsion system engine and defines the test utilization of the hardware enumerated in Table 5-1. The specific test levels to be applied in


HELIUM TANK
HELIUM TANK
HELIUM FILL AND DRAIN VALVE
CAP
SOLENOID VALVE N.O.
SOLENOID VALVE N.O.
REGULATOR
REGULATOR
REGULATOR
REGULATOR
CHECK VALVE
CHECK VALVE
CHECK VALVE
CHECK VALVE
BURST DISC
CHECK VALVE
CHECK VALVE
CHECK VALVE
CHECK VALVE
FUEL RELIEF VALVE
BURST DISC
FILTER
FUEL VENT VALVE
CAP
FUEL TANK
FUEL TANK
HEAT - EXCHANGER
CHECK VALVE ACTUATOR VENT
CHECK VALVE ACTUATOR VENT
CHECK VALVE ACTUATOR VENT
CHECK VALVE ACTUATOR VENT
INJECTOR CHAMBER AND NOZZLE EXTENSION
GIMBAL BEARING
GIMBAL ACTUATOR
GIMBAL ACTUATOR
GIMBAL ACTUATOR
GIMBAL ACTUATOR
PROPELLANT UTILIZATION SYSTEM

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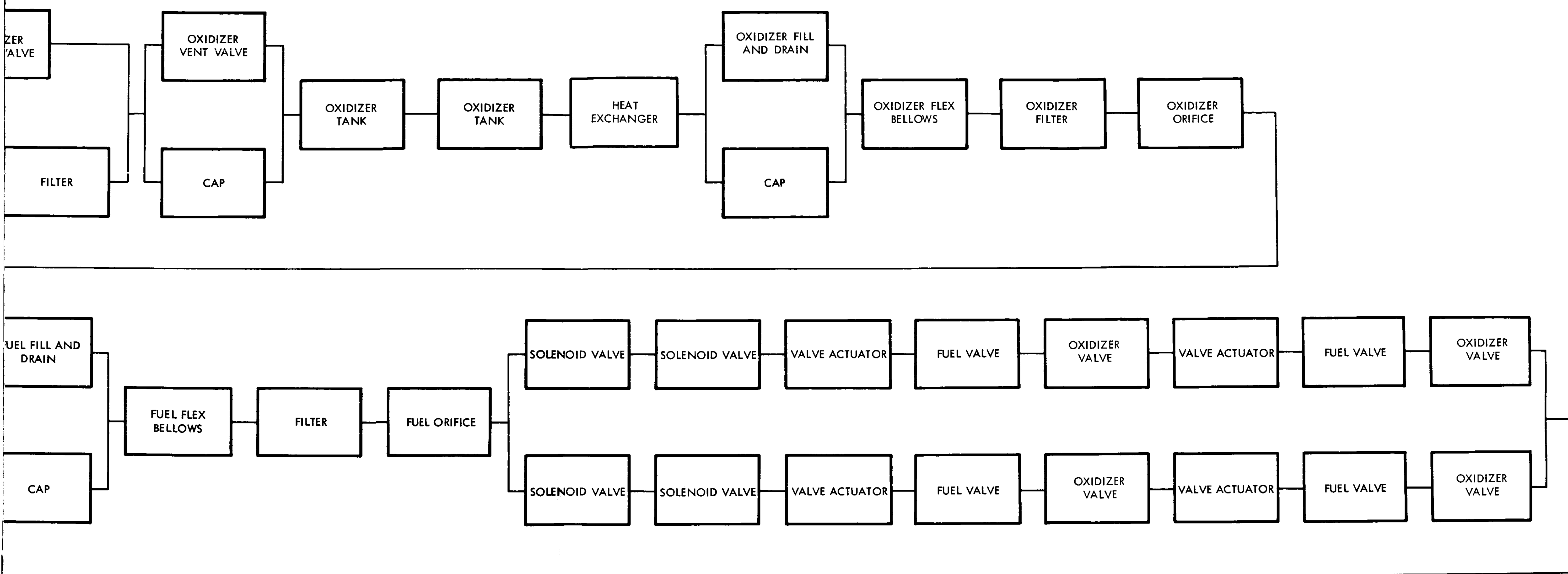

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Figure 5-1. SPS Reliability Assessment Model

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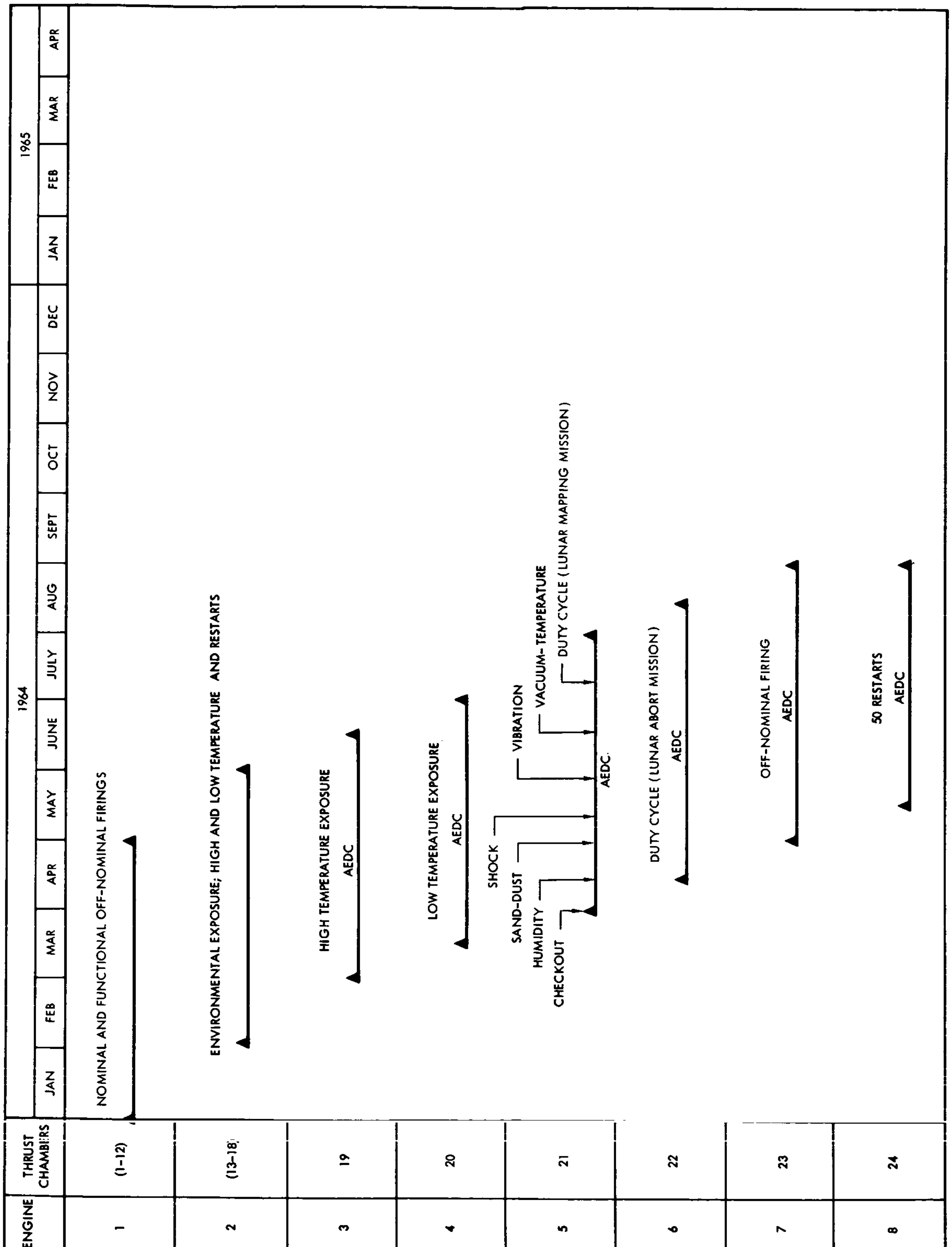

Figure 5-2. SPS Engine Test Schedule

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qualification testing are found in the tables of environmental test criteria in subsection 4.2 and are a function of the zone location of the equipment in the spacecraft. The zone number is given in Table 5-1.

Table 5-2. SPS Propellant Feed Parameters for Qualification Testing

Component	S/C Zone	Criticality	Required Hardware	Qualification Test Phase Distribution	
				Qual	Off-Limit
Heat exchangers	3	1	4	4	2
Disconnects	3	3	4	4	2
Connectors	3	1	4	4	2
Valves	3	3	4	4	2
Regulators	3	3	4	4	2
Quantity Gaging and Indicator Subsystem	1	1	4	4	2
Tanks (3 types)	1	1	4	4	2

5.1.4.2 Propellant Feed

Figure 5-3 displays the qualification test schedules for the SPS propellant feed components and defines the test utilization of the hardware enumerated in Table 5-2. The specific test levels to be applied in qualification testing are found in the tables of environmental test criteria in subsection 4.2 and are a function of the zone location of the equipment in the spacecraft. The zone number is given in Table 5-2.

5.2 REACTION CONTROL SYSTEM

5.2.1 Scope

Qualification tests on the reaction control subsystem engines conform to the general requirements of MIL-E-5151 and include the following:

1. Calibration tests

2. Environmental tests

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[redacted] [redacted]

COMPONENTS	ITEM NO.	WEEK NO. 1–17 (test sequence)
HEAT EXCHANGER, OXIDIZER; HEAT EXCHANGER, FUEL	1 - 4	1, 2, 6, 8, 9, 17, 12
VALVE, CHECK	1	1, 18, 6, 6, 12
	2	1, 18, 6, 12
	3	1, 6, 17, 16, 12
	4	1, 6, 17, 16, 12
VALVE, RELIEF, HELIUM	1	1, 2, 6, 5, 10, 6, 12
	2	1, 2, 6, 5, 10, 17, 12, 16
	3	1, 6, 13, 6, 16, 12
	4	1, 6, 13, 16, 12
REGULATOR, PRESSURE, HELIUM	1	1, 6, 5, 8, 9, 16, 12
	2	1, 6, 5, 8, 9, 16, 12
	3	1, 13, 15, 16
	4	1, 13, 15, 16
VALVE, SOLENOID, HELIUM	1	1, 3, 4, 6, 5, 8, 9, 16, 12
	2	1, 3, 4, 6, 5, 8, 9, 6, 12
	3	1, 15, 8, 9, 16
	4	1, 15, 8, 9, 16
DISCONNECT, FILL, HELIUM; DISCONNECT, TEST, HELIUM; DISCONNECT, FILL, OXIDIZER; DISCONNECT, TEST, OXIDIZER; DISCONNECT, VENT, OXIDIZER; DISCONNECT, FILL, FUEL; DISCONNECT, TEST, FUEL; DISCONNECT, VENT, FUEL — GROUND HALF	1 & 2	1, 2, 18, 11, 16, 12
	3 & 4	1, 18, 16, 2, 17
AIRBORNE HALF	1 & 2	1, 2, 18, 6, 10, 16, 6, 12
	3 & 4	1, 18, 16, 2, 17
CONNECTOR, FUEL; CONNECTOR, OXIDIZER	1 - 4	1, 18, 2, 6, 12
SPS QUANTITY GAGING AND INDICATOR SUBSYSTEM	1 - 4	1, 18, 2, 16, 11, 7, 6, 5, 10, 12

LEGEND

1 ACCEPTANCE
2 HUMIDITY
3 ELECTROMAGNETIC INTERFERENCE
4 EXPLOSIVE ATMOSPHERE
5 ACCELERATION
6 VIBRATION
7 ACOUSTICS
8 TEMPERATURE, HIGH
9 TEMPERATURE, LOW
10 VACUUM
11 SHOCK
12 BURST PRESSURE
13 VACUUM, TEMPERATURE
14 VIBRATION, ACCELERATION
15 VIBRATION, TEMPERATURE
16 PERFORMANCE
17 SURGE PRESSURE
18 FLUID COMPATIBILITY

Figure 5-3. SPS Propellant Feed Components (Less Tanks) Qualification Test Schedule (Sheet 1 of 2)

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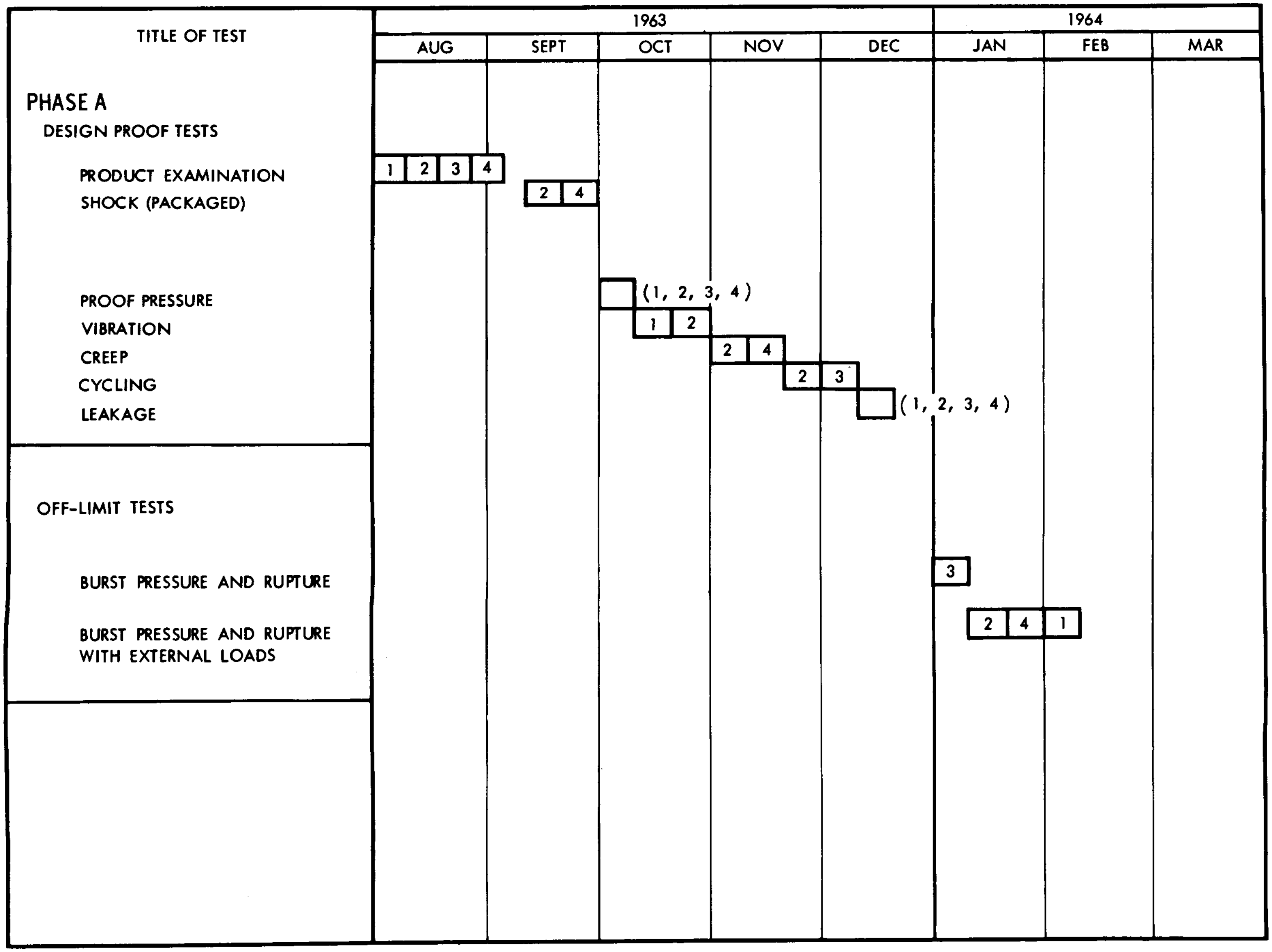


Figure 5-3. SPS Propellant Feed Components (Less Tanks) Qualification Test Schedule (Sheet 2 of 2)

3. Electromagnetic interference and susceptibility tests

4. Mission simulation life tests

The number of test engines is predicated on the requirement to assess compliance with the procurement specification in normal and off-limit modes of operation. The number of qualification test samples for the components of the propellant feed subsystem is based on the criticality index, limited life requirements, service life considerations, and the multiple use of the same test articles.

5.2.2 Reliability Assessment Model

5.2.2.1 Command Module

Figure 5-4 depicts the reliability assessment model for the C/M RCS. The model will aid in establishing test requirements and in programming the collection of data for the assessment of reliability by illustrating the interrelationship of system components.

5.2.2.2 Service Module

Figure 5-5 depicts the reliability assessment model for the S/M RCS. The model will aid in establishing test requirements and in programming the collection of data for the assessment of reliability by illustrating the interrelationship of system components.

5.2.3 Criticality and Hardware Requirements

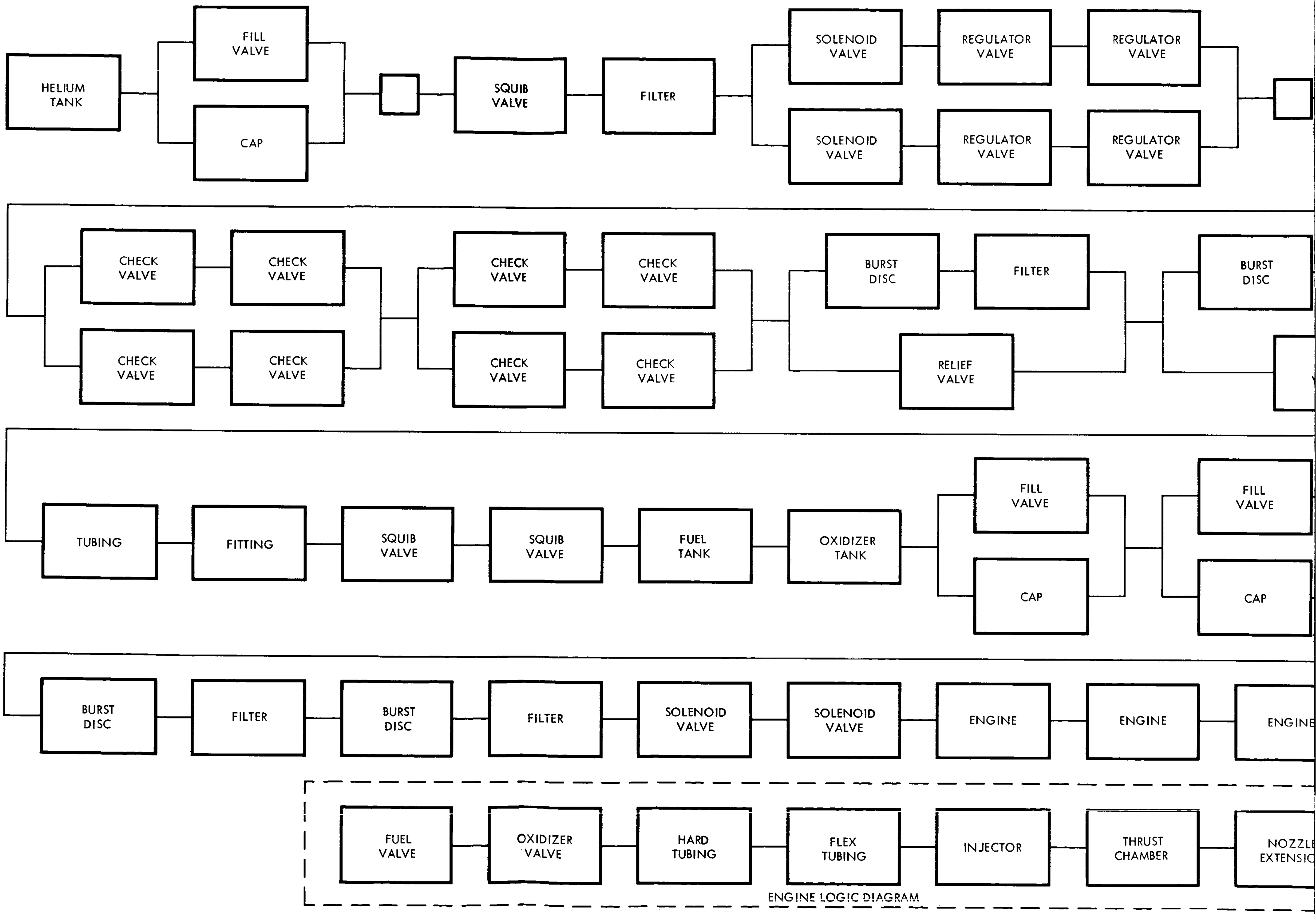
5.2.3.1 Command Module

5.2.3.1.1 Engine. Table 5-3 designates the criticality class, as defined in subsection 2.4, for the components of the C/M rocket engine. The table enumerates the hardware requirements for a minimum qualification program as established in subsection 2.5. The location of the components in the spacecraft is indicated by spacecraft zone numbers as defined in subsection 4.2.

Table 5-3. C/M RCS Engine Parameters for Qualification Testing

Component	S/C Zone	Criticality	Required Hardware
Thrust chamber	1	2	16
Injector	1	2	14
Fuel valve	1	2	14
Oxidizer valve	1	2	14

[redacted]



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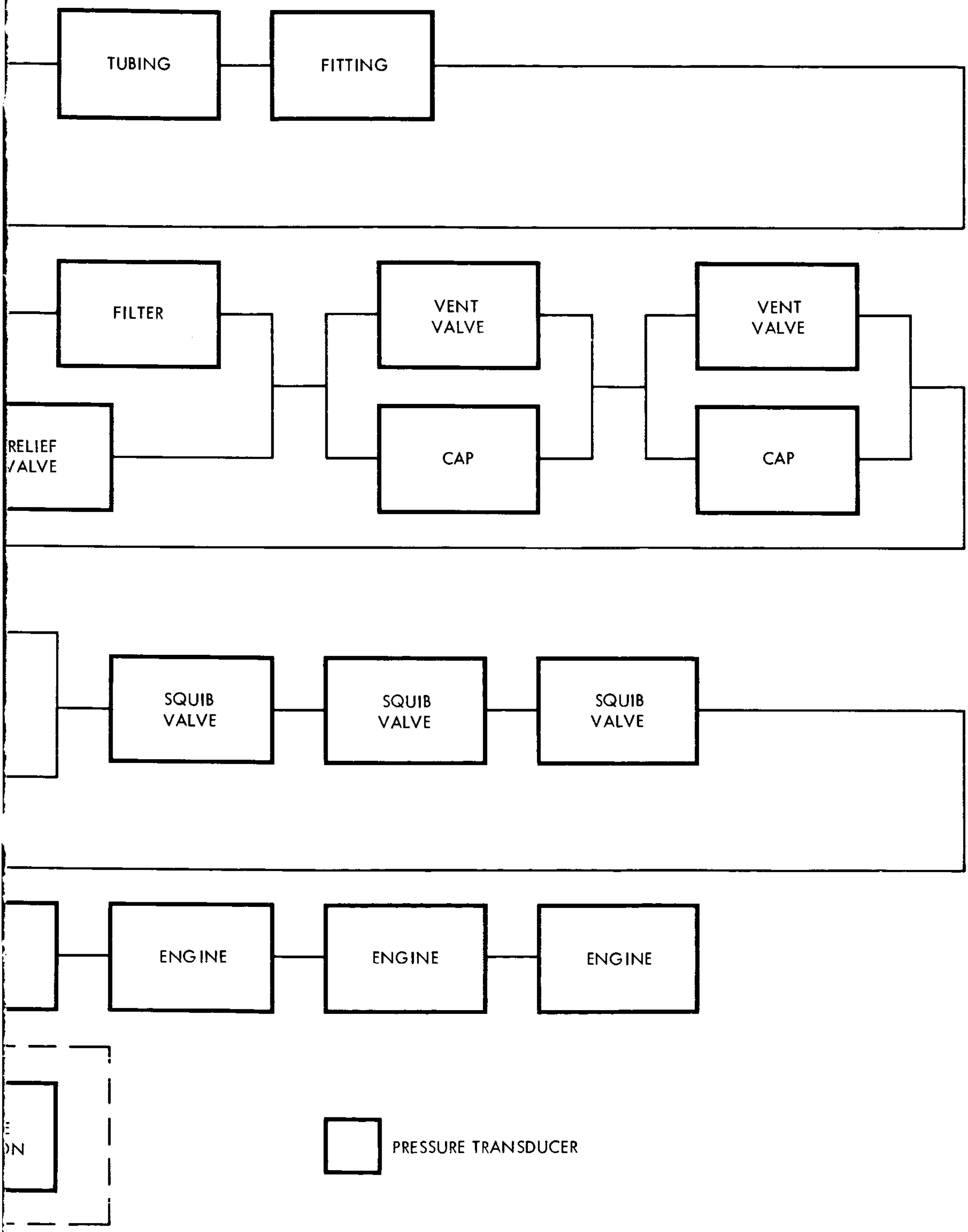

Figure 5-4. C/M RCS Reliability Assessment Model

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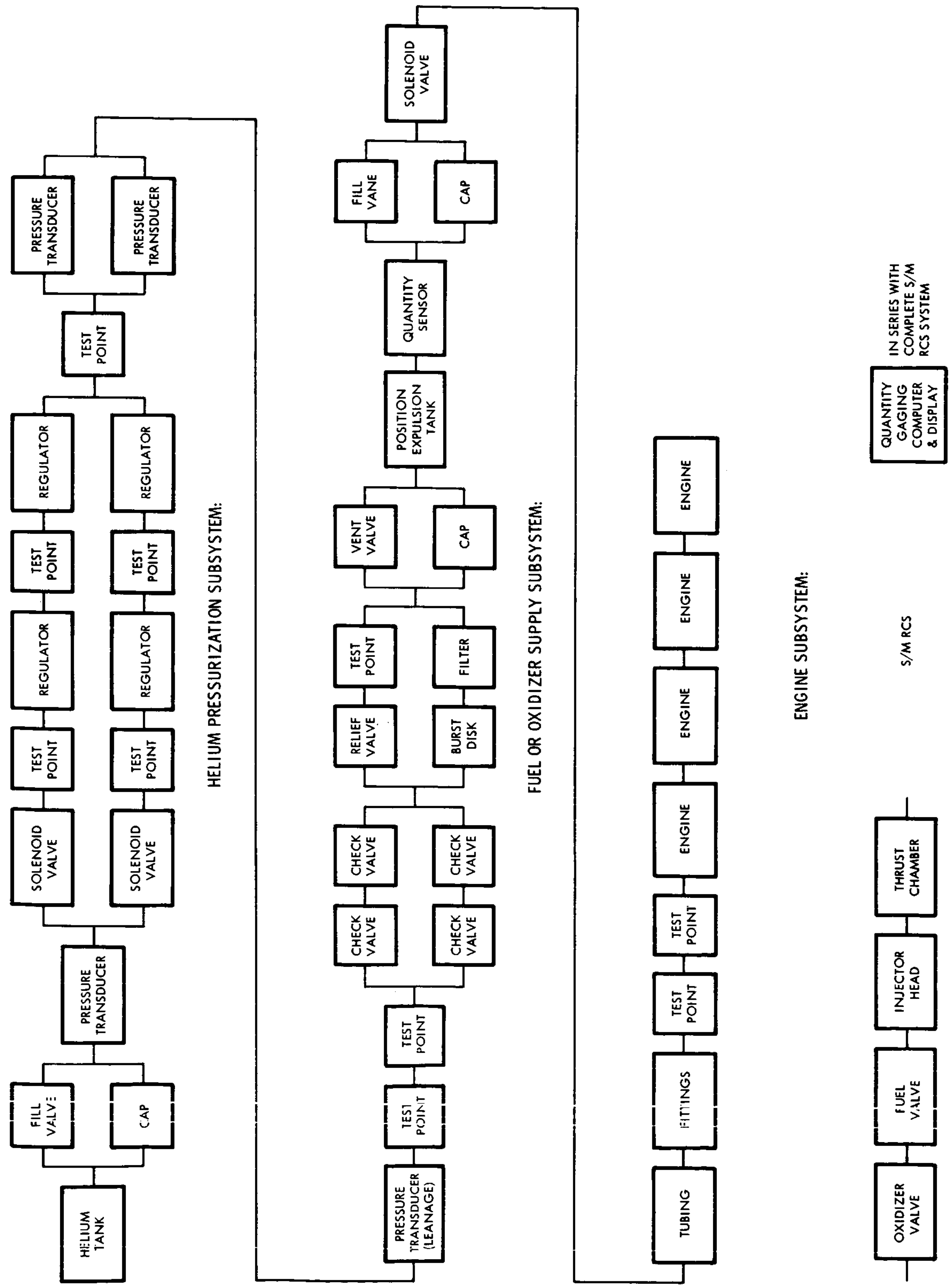

Figure 5-5. S/M RCS Module Configuration

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5.2.3.1.2 Propellant Feed. Table 5-4 designates the criticality class, as defined in Section 2.4, for the components of the C/M RCS Propellant Feed. The table enumerates the hardware requirements for a minimum qualification program, as established in Section 2.5. The distribution of the hardware through the subsystem qualification test phases, as described in Section 2.3, is shown. The location of the components in the spacecraft is indicated by spacecraft zone numbers, as defined in Section 4.2.

Table 5-4. C/M RCS Propellant Feed Parameters for Qualification Testing

Component	S/C Zone	Criticality	Required Hardware	Qualification Test Phase Distribution		
				A-1	A-2	B
Burst Disc	1	2	4	2	1	1
Squib valve	1	2	5	2	1	2
Disconnect	1	3	6	2	1	3
Valves, solenoid check or relief	1	3	8	2	2	4
Regulator	1	3	4	1	1	2
Tanks (3 types)	1	2	6	6	6	0

5.2.3.2 Service Module

5.2.3.2.1 Engine. Table 5-5 designates the criticality class, as defined in subsection 2.4, for the components of the S/M rocket engine. The table enumerates the hardware requirements for a minimum qualification program as established in subsection 2.5. The location of the components in the spacecraft is indicated by spacecraft zone numbers as defined in subsection 4.2.

Table 5-5. S/M RCS Rocket Engine Parameters for Qualification Testing

Component	S/C Zone	Criticality	Required Hardware
Thrust chamber	1	3	13
Injector	1	3	9
Fuel valve	1	3	10
Oxidizer valve	1	3	10

[redacted]IAL

[redacted] [redacted]

5.2.3.2.2 Propellant Feed. Table 5-6 designates the criticality class, as defined in subsection 2.4, for the components of the S/M RCS propellant feed. The table enumerates the hardware requirements for a minimum qualification program as established in Section 2.5. The distribution of the hardware through the subsystem qualification test phases, as described in subsection 2.3, is shown. The location of the components in the spacecraft is indicated by spacecraft zone numbers, as defined in subsection 4.2.

Table 5-6. S/M RCS Propellant Feed Parameters for Qualification Testing

Component	S/C Zone	Criticality	Required Hardware	Qualification Test Phase Distribution		
				A-1	A-2	B
Disconnect	1	3	6	2	1	3
Flex hose	1	3	4	1	1	2
Regulator	1	3	6	2	1	3
Quantity gaging	1	3	3	1	1	1
Valve	1	3	4	1	1	2
Tanks (3 types)	1	3	5	5	5	0

5.2.4 Qualification Test Schedule

5.2.4.1 Command Module

5.2.4.1.1 Engine. Figure 5-6 displays the qualification test schedule for the C/M RCS engines and defines the test utilization of the hardware enumerated in Table 5-3. The specific test levels to be applied in qualification testing are found in the tables of environmental test criteria in subsection 4.2 and are a function of the zone location of the equipment in the spacecraft. These zone numbers are given in Table 5-3.

5.2.4.1.2 Propellant Feed. Figure 5-7 displays the qualification test schedule for the C/M RCS propellant feed and defines the test utilization of the hardware enumerated in Table 5-4. The specific test levels to be applied in qualification testing are found in the tables of environmental test criteria in subsection 4.2 and are a function of the zone location of the equipment in the spacecraft. These zone numbers are given in Table 5-4.

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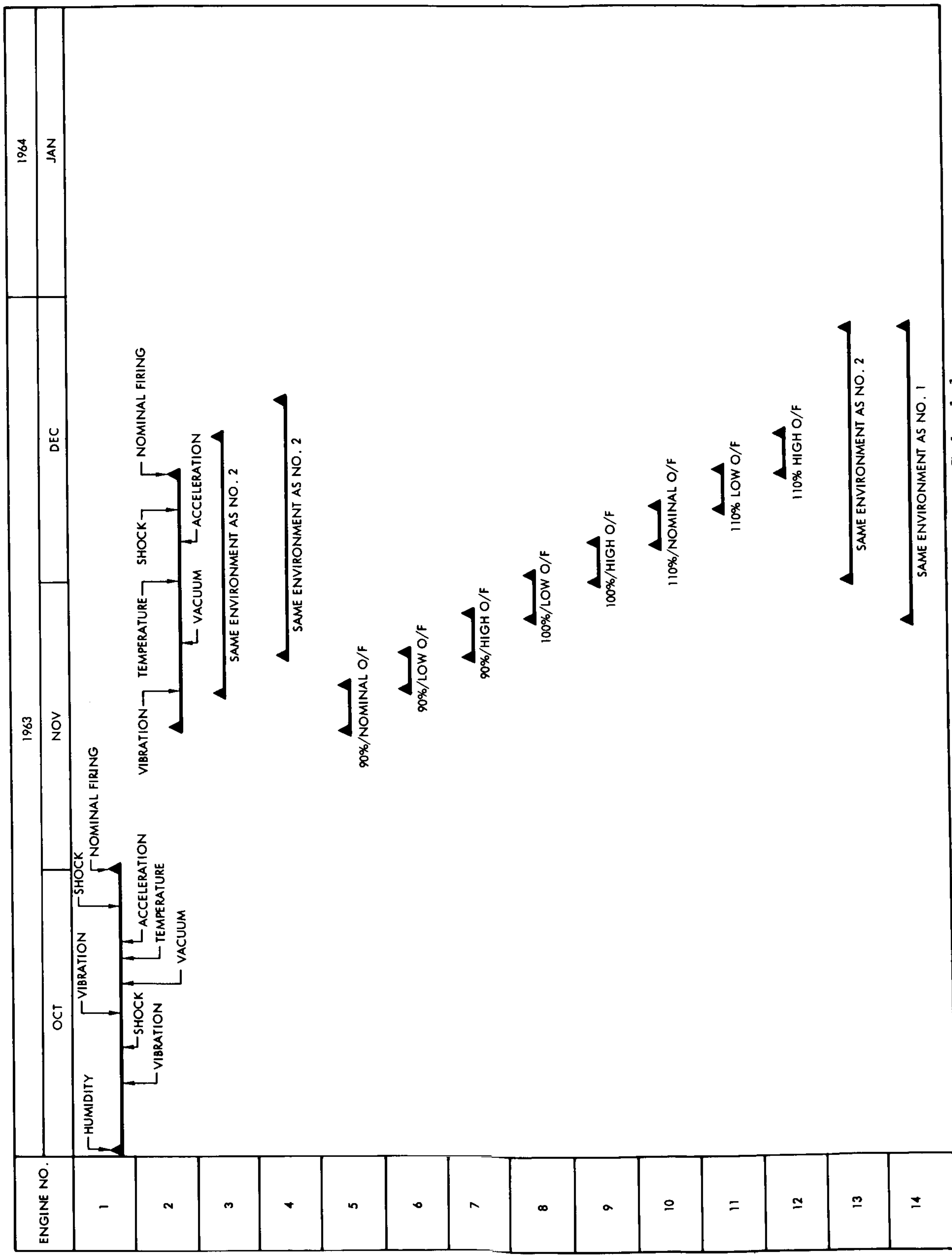

Figure 5-6. C/M RCS Engine Test Schedule

[redacted] [redacted]

COMPONENTS	ITEM NO.	TEST SEQUENCE (WEEK NO. 1–15)
VALVE, RELIEF, HELIUM	1	1, 6, 16, 19, 20, 6, 12
	2	1, 16, 12
	3	1, 6, 16, 19, 20
	4	1, 12, 19, 20
VALVE, SOLENOID, HELIUM VALVE, SOLENOID, OXIDIZER VALVE, SOLENOID, FUEL	1	1, 3, 4, 5, 16, 6, 1, 12, 20
	2	1, 3, 4, 5, 16, 6, 1, 12, 20
	3	1, 3, 5, 6, 16, 1, 20
	4	1, 3, 5, 6, 16, 1, 20
VALVE, SQUIB, HELIUM	1	1, 2, 3, 6, 8, 9, 4
	2	1, 2, 3, 6, 8, 9, 6, 4, 12
	3	1, 2, 3, 6, 8, 9, 6, 4, 12
	4	1, 15, 13, 16
	5	1, 15, 13, 6, 16, 12
DISC, BURST, OXIDIZER DISC, BURST, FUEL	1	1, 18, 6, 16, 12
	2	1, 18, 6, 16, 12
	3	1, 18, 16, 12
	4	1, 18, 16, 12
COMPONENT: REGULATOR, PRESSURE, HELIUM SAMPLE:	1	1, 19, 16, 6, 16
	2	1, 19, 16, 6, 16, 20
	3	1, 19, 16, 6, 13
	4	1, 19, 16, 6, 13, 20
C/M RCS REGULATOR COMPONENT: VALVE, CHECK SAMPLE:	1	1, 19, 16, 6, 16, 19, 20
	2	1, 19, 16, 6, 16, 19, 20
	3	1, 19, 16, 6, 16, 19, 20
	4	1, 19, 16, 6, 19, 20
COMPONENTS: 1. DISCONNECT, FILL, HELIUM 2. DISCONNECT, TEST, HELIUM 3. DISCONNECT, FILL, OXIDIZER 4. DISCONNECT, TEST, OXIDIZER 5. DISCONNECT, VENT, OXIDIZER 6. DISCONNECT, FILL, FUEL 7. DISCONNECT, TEST, FUEL 8. DISCONNECT, VENT, FUEL SAMPLE: (AIRBORNE HALF)	1	1, 16, 11, 6, 16
	2	1, 16, 6, 16
	3	1, 16, 6
	4	1, 16, 6
SAMPLE: (GROUND HALF)	1	1, 16, 11, 6, 16
	2	1, 16, 6, 16
	3	1, 16, 6
	4	1, 16, 6

LEGEND

1 ACCEPTANCE
2 HUMIDITY
3 ELECTROMAGNETIC INTERFERENCE
4 EXPLOSIVE ATMOSPHERE
5 ACCELERATION
6 VIBRATION
8 TEMPERATURE, HIGH
9 TEMPERATURE, LOW
10 VACUUM
11 SHOCK
12 BURST PRESSURE
13 VACUUM, TEMPERATURE
14 VIBRATION, ACCELERATION
15 VIBRATION, TEMPERATURE
16 PERFORMANCE
17 SURGE PRESSURE
18 FLUID COMPATABILITY
19. CLEANLINESS
20. DISASSEMBLY AND INSPECTION

Figure 5-7. C/M RCS Propellant Feed (Less Tanks) Qualification Test Schedule (Sheet 1 of 2)

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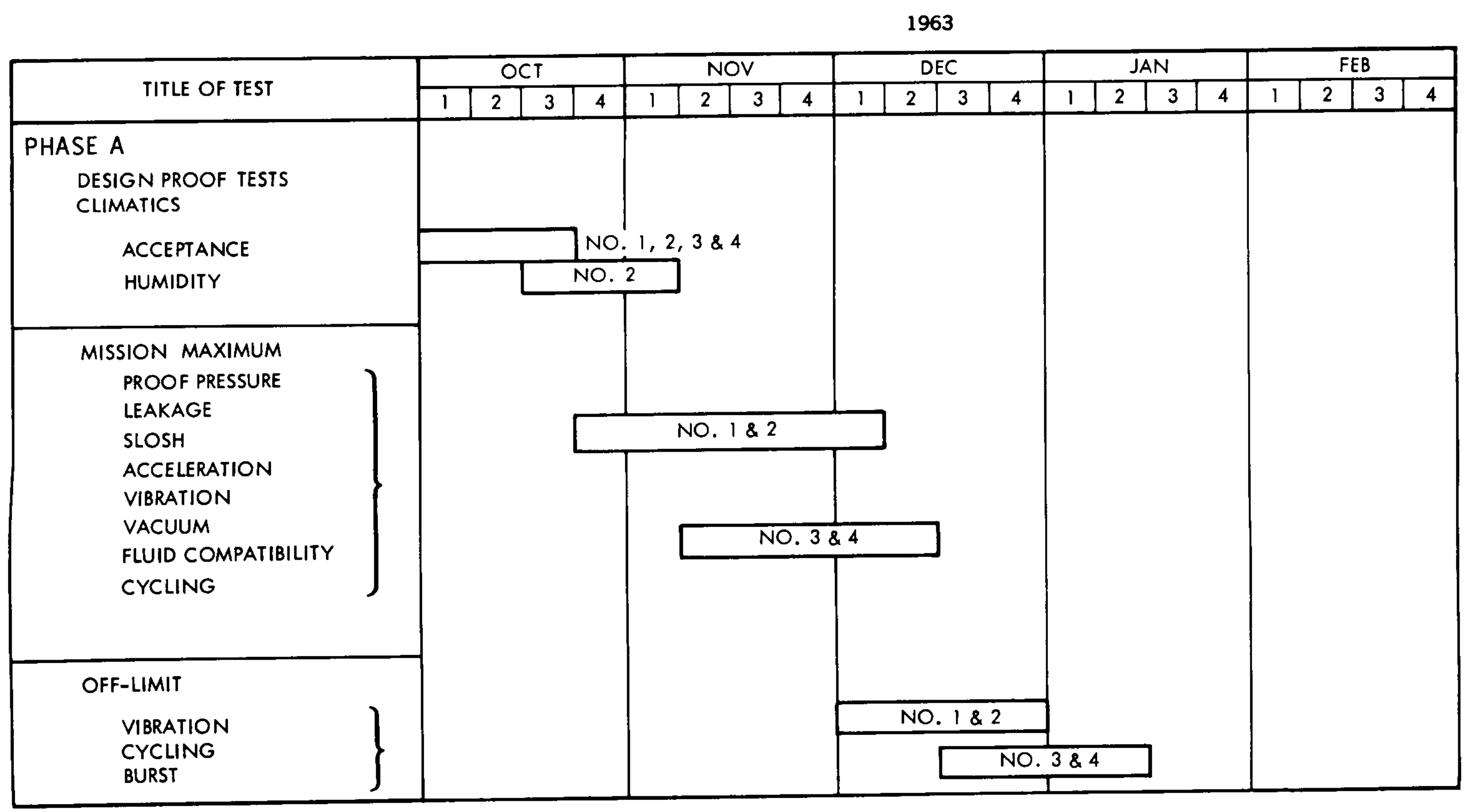

Figure 5-7. C/M RCS Propellant Feed (Tanks) Qualification Test Schedule (Sheet 2 of 2)

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5.2.4.2 Service Module

5.2.4.2.1 Engine. Figure 5-8 displays the qualification test schedule for the S/M RCS engines and defines the test utilization of the hardware enumerated in Table 5-5. Qualification tests will use eight complete engines and four replacement thrust chambers. The thirteenth thrust chamber will be used for component testing. The specific test levels to be applied in qualification testing are found in the tables of environmental test criteria in subsection 4.2 and are a function of the zone location of the equipment in the spacecraft. These zone numbers are given in Table 5-5.

5.2.4.2.2 Propellant Feed. Figure 5-9 displays the qualification test schedules for the S/M RCS propellant feed components and defines the test utilization of the hardware enumerated in Table 5-6. The specific test levels to be applied in qualification testing are found in the tables of environmental test criteria in subsection 4.2 and are a function of the zone location of the equipment in the spacecraft. These zone numbers are given in Table 5-6.

5.3 LAUNCH ESCAPE SYSTEM

5.3.1 Scope

Qualification testing of launch escape, tower jettison, and pitch control motors will be accomplished on representative production samples of units and associated components to determine compliance with functional and performance specification requirements. The number of test motors is based on the requirements deemed necessary to assess compliance with the specification. The test program will subject the equipment to a series of environments as anticipated in service.

5.3.2 Reliability Assessment Model

Figure 5-10 depicts the reliability assessment model for the LES. The model will aid in establishing test requirements and in programming the collection of data for the assessment of reliability by illustrating the interrelationship of system components.

5.3.3 Criticality and Hardware Requirements

Table 5-7 designates the criticality class, as defined in subsection 2.4, for the LES rocket motors. The table enumerates the hardware requirements for an optimum qualification program as established in subsection 2.5.

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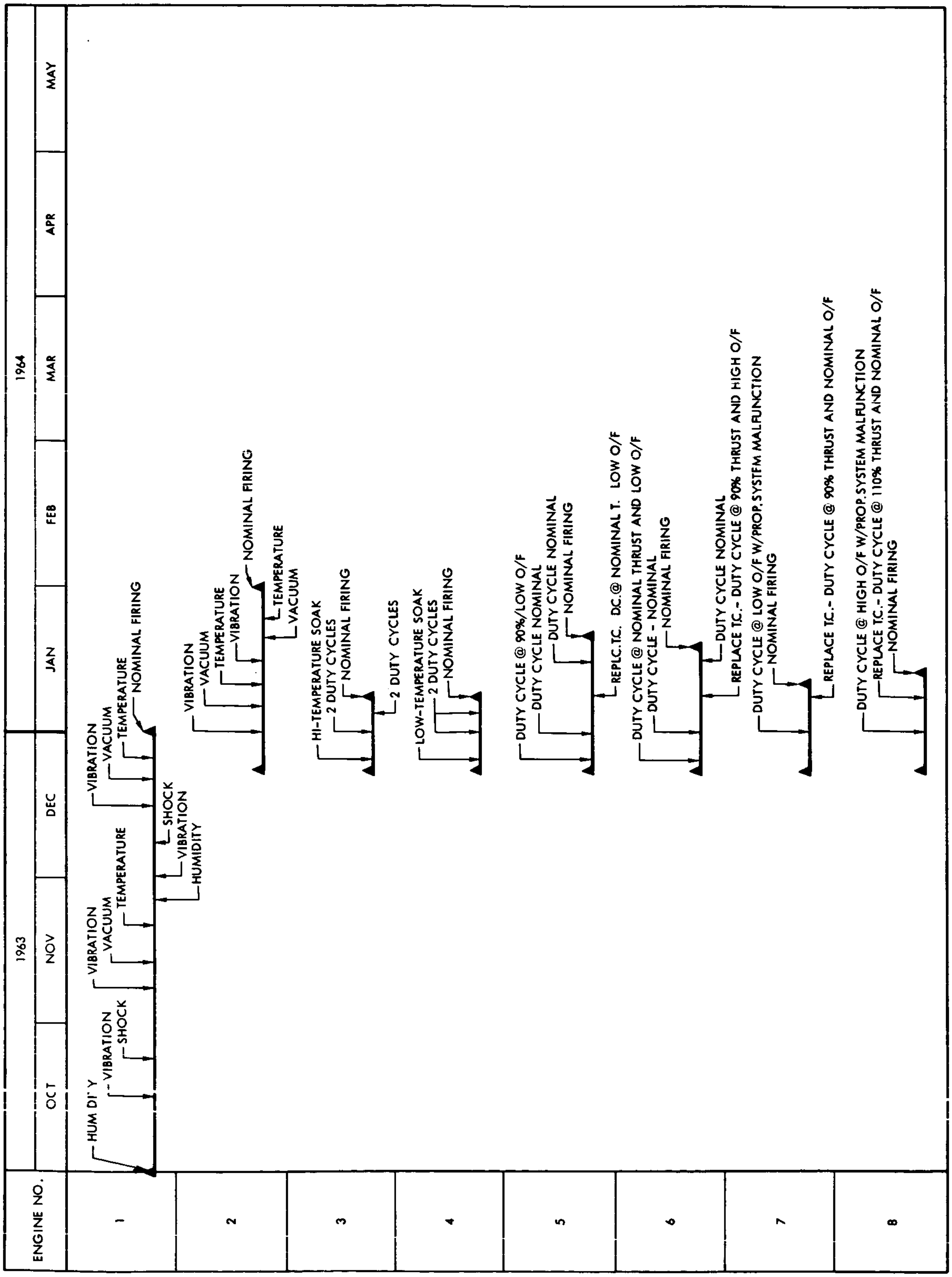

Figure 5-8. S/M RCS Engine Test Schedule

[redacted]

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COMPONENTS	ITEM NO.	TEST SEQUENCE (WEEK NO. 1–17)
FLEX HOSE	1	1, 6, 15, 16, 18, 12
	2	1, 6, 9, 16, 18, 12
	3	1, 6, 15, 16, 17, 12
	4	1, 6, 9, 16, 12
DISCONNECT, FILL, HELIUM; DISCONNECT, TEST, HELIUM; DISCONNECT, FILL, OXIDIZER; DISCONNECT, TEST, OXIDIZER; DISCONNECT, VENT, OXIDIZER; DISCONNECT, FILL, FUEL; DISCONNECT, TEST, FUEL; DISCONNECT, VENT, FUEL — GROUND HALF	1	1, 16, 11, 6, 16
	2	1, 16, 6, 16
	3	1, 16, 6
	4	1, 16, 6
AIRBORNE HALF	1	1, 16, 11, 6, 16
	2	1, 16, 6, 16
	3	1, 16, 6
	4	1, 16, 6
VALVE, SOLENOID, HELIUM; VALVE, SOLENOID, FUEL; VALVE, SOLENOID, OXIDIZER	1	1, 3, 4, 5, 16, 6, 1, 12, 20
	2	1, 3, 4, 5, 16, 6, 1, 12, 20
	3	1, 3, 5, 6, 16, 1, 20
	4	1, 3, 5, 6, 16, 1, 20
S/M RCS SOLENOID VALVES; REGULATOR, PRESSURE, HELIUM	1	1, 19, 16, 6, 16
	2	1, 19, 16, 6, 16, 20
	3	1, 19, 16, 6, 13
	4	1, 19, 16, 6, 13, 20
S/M RCS REGULATOR; VALVE, RELIEF, HELIUM	1	1, 6, 16, 19, 20, 6, 12
	2	1, 16, 12
	3	1, 6, 16, 19, 20
	4	1, 12, 19, 20
S/M RCS RELIEF VALVE; VALVE, CHECK	1	1, 19, 16, 6, 16, 19, 20
	2	1, 19, 16, 6, 16, 19, 20
	3	1, 19, 16, 6, 16, 19, 20
	4	1, 19, 16, 6, 19, 20
PROPELLANT QUANTITY GAUGING	1	1, 3, 4, 6, 10, 1, 15, 16
	2	1, 3, 4, 6, 10, 1, 16
	3	1, 3, 4, 6, 8, 11, 15, 16

LEGEND

1 ACCEPTANCE
2 HUMIDITY
3 ELECTROMAGNETIC INTERFERENCE
4 EXPLOSIVE ATMOSPHERE
5 ACCELERATION
6 VIBRATION
8 TEMPERATURE, HIGH
9 TEMPERATURE, LOW
10 VACUUM
11 SHOCK
12 BURST PRESSURE
13 VACUUM, TEMPERATURE
14 VIBRATION, ACCELERATION
15 VIBRATION, TEMPERATURE
16 PERFORMANCE
17 SURGE PRESSURE
18 FLUID COMPATIBILITY
19 CLEANLINESS
20 DISASSEMBLY & INSPECTION

Figure 5-9. S/M RCS Propellant Feed (Less Tanks) Qualification Test Schedule (Sheet 1 of 2)

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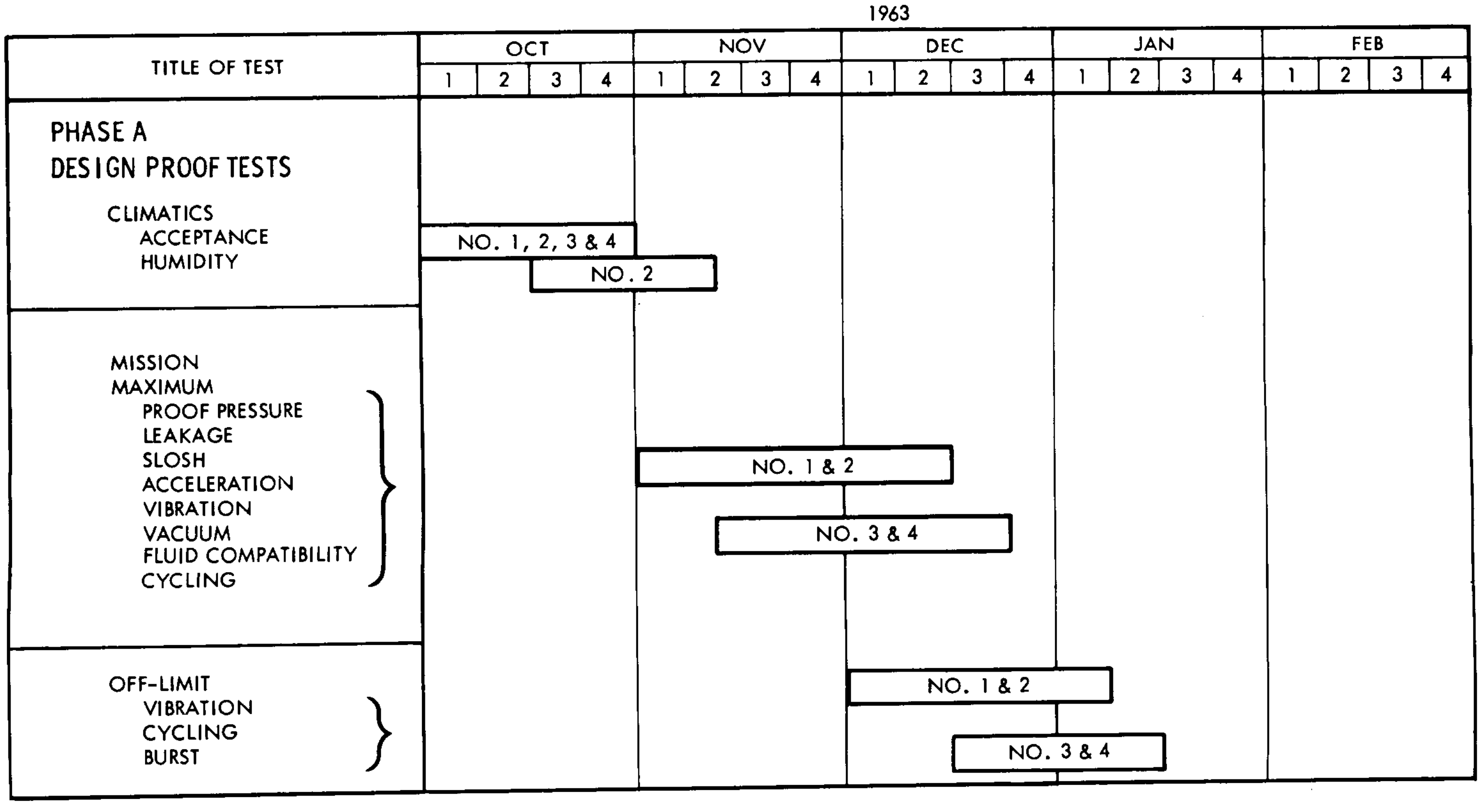

Figure 5-9. S/M RCS Propellant Feed (Tanks) Qualification Test Schedule (Sheet 2 of 2)

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[redacted] [redacted]

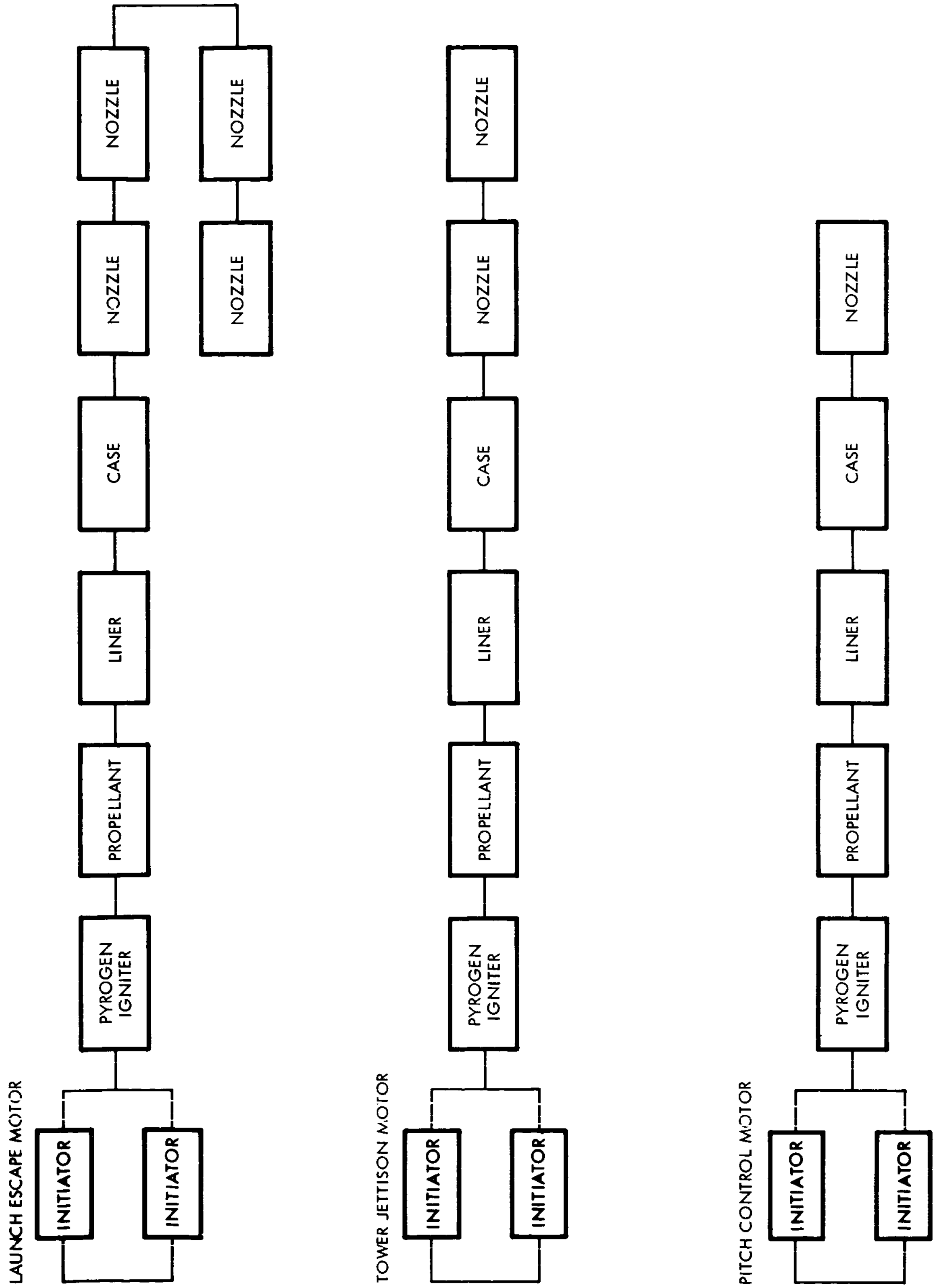

Figure 5-10. LES Reliability Assessment Model

[redacted]

[redacted] [redacted]

Table 5-7. LES Rocket Motor Criticality and Hardware Requirements

Components	Criticality	Number of Units Required
Tower jettison motor	2	28
Launch escape motor	2	28
Pitch control motor	3	30
Igniter cartridge, hot wire	1	118

NOTE: The additional two pitch control motors are required for acceleration tests. See Table 5-9.

5.3.4 Qualification Test Schedule

5.3.4.1 Tower Jettison Motor

Figure 5-11 displays the qualification test schedule for the LES tower jettison motor. Table 5-8 programs the test articles enumerated in Table 5-7 in accordance with the test schedule. Since the test utilization of the launch escape motor hardware is similar to that of the tower jettison motor, Table 5-8 is also applicable to the launch escape motor.

5.3.4.2 Launch Escape Motor

Figure 5-12 displays the qualification test schedule for the LES launch escape motor. The schedule defines the test utilization of the hardware enumerated in Table 5-7 in accordance with the test program of Table 5-8.

5.3.4.3 Pitch Control Motor

Figure 5-13 displays the qualification test schedule for the LES pitch control motor. The schedule defines the test utilization of the hardware enumerated in Table 5-7 in accordance with the test program of Table 5-9.

5.3.4.4 Hot Wire Igniter Cartridge

Table 5-10 defines the test utilization of the hardware enumerated in Table 5-7. The specific test levels to be applied in qualification testing are found in the tables of environmental test criteria in Section 4.2.

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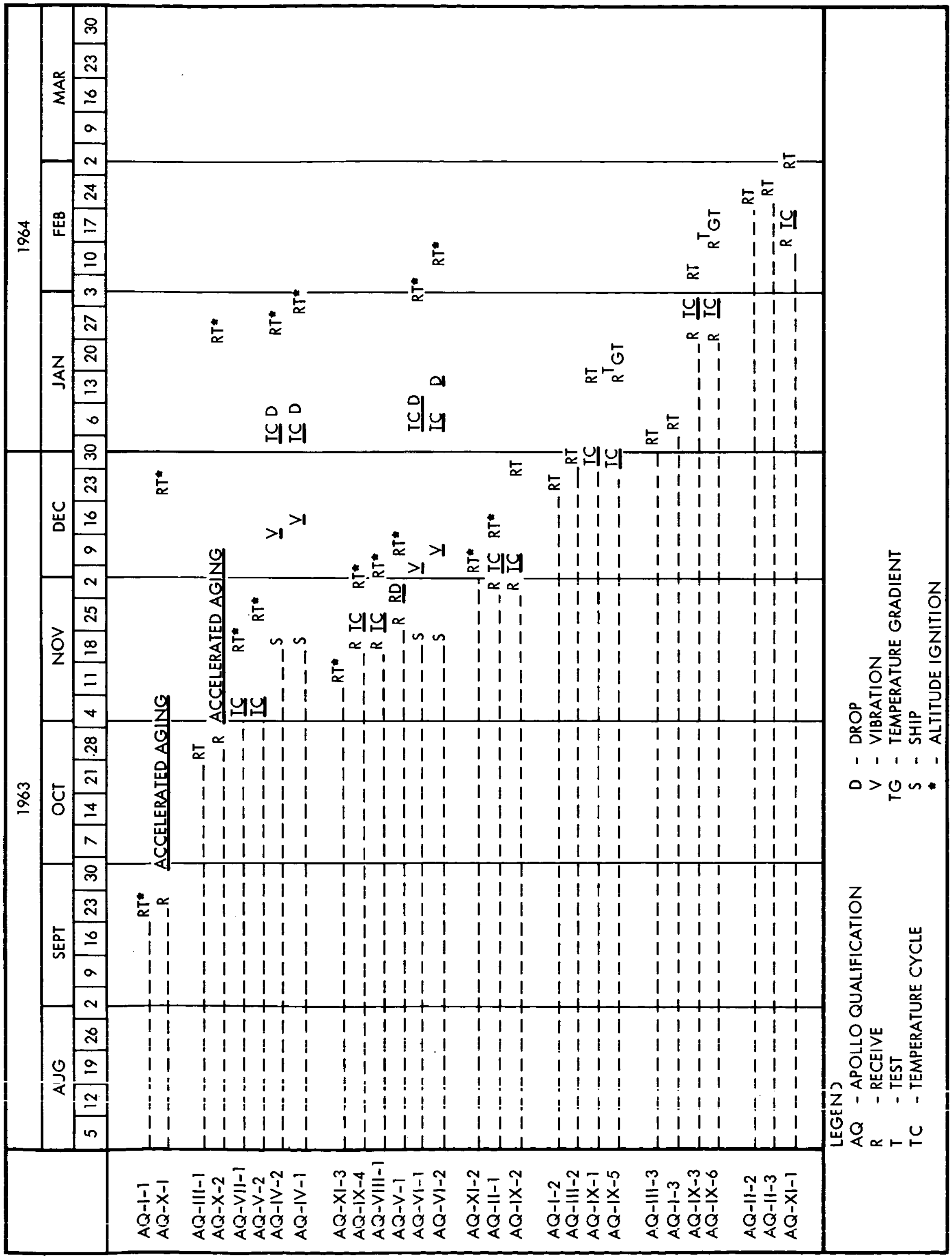

Figure 5-11. LES Tower Jettison Motor Qualification Test Schedule

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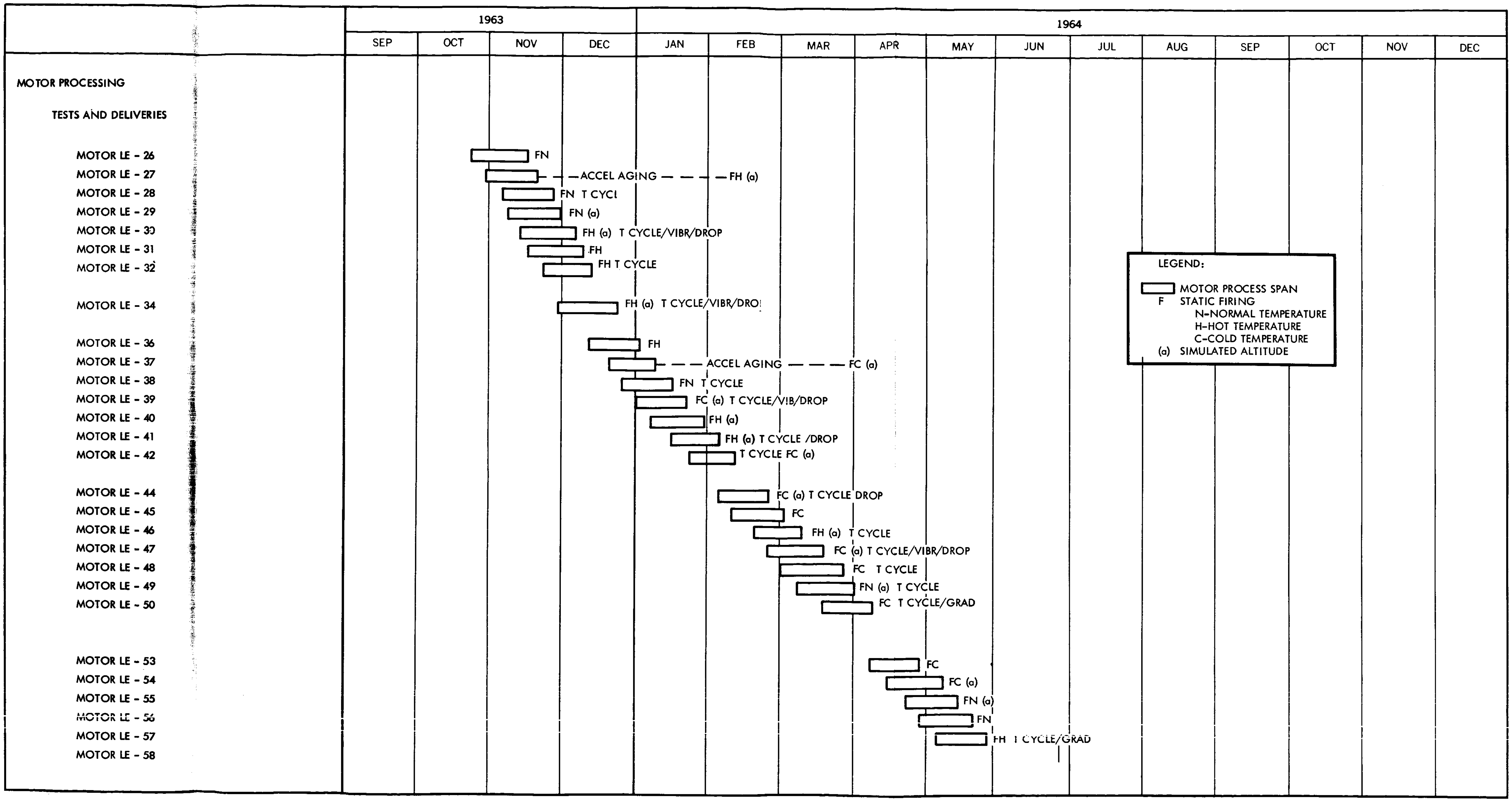

Figure 5-12. Launch Escape Motor Qualification Test Schedule

5-27, 5-28

[redacted]

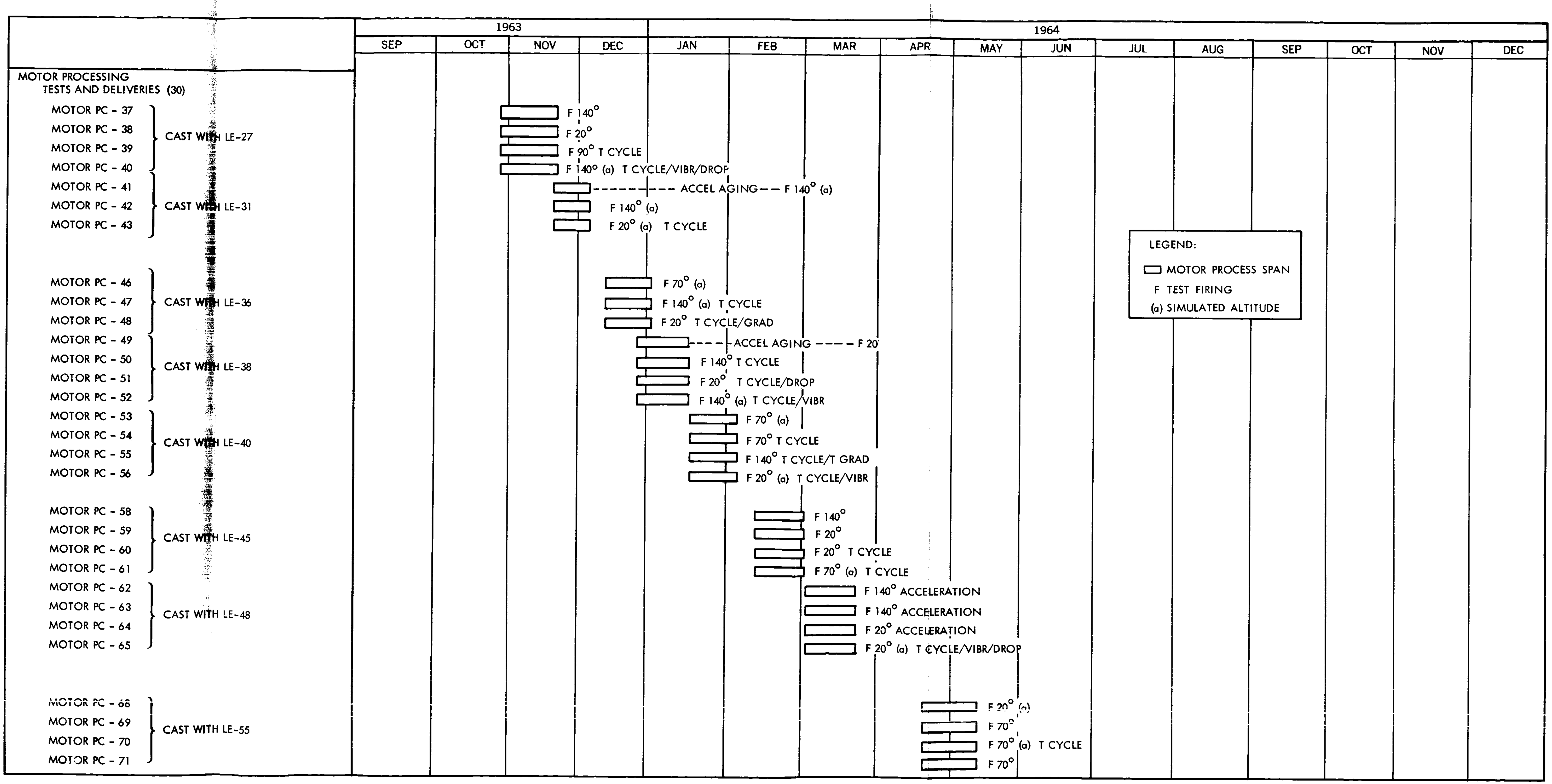

Figure 5-13. Pitch Control Motor Qualification Test Schedule

[redacted] [redacted]

Table 5-8. Program for Test Articles, Tower Jettison and Launch Escape Motors

	Test Group and Number of Motors										
	I (3)	II (3)	III (3)	IV (2)	V (2)	VI (2)	VII (1)	VIII (1)	IX (6)	X (2)	XI (3)
Statistical firing											
High temperature	2								1		
Normal temperature		2							1		1
Low temperature			2						1		
Sequential treatment											
Temperature cycle		1		2	2	2	1	1	6		1
Vibration											
High temperature				2							
Low temperature						2					
Drop											
High temperature				2							
Normal temperature					2						
Low temperature						2					
Accelerated age										2	
Altitude firing											
High temperature	1			1	1	1		1		1	
Normal temperature		1							1		2
Low temperature			1	1	1	1	1			1	
Temperature gradient											
High to low									1		
Low to high									1		

[redacted]

Table 5-9. Program for Test Articles, Pitch Control Motor

	Test Group and Number of Motors											
	I (3)	II (3)	III (3)	IV (2)	V (1)	VI (2)	VII (1)	VIII (1)	IX (6)	X (2)	XI (3)	XII (3)
Statistical firing												
High temperature	2								1			
Normal temperature		2							1		1	
Low temperature			2						1			
Sequential temperature												
Temperature cycle		1		2	1	2	1	1	1		1	
Vibration												
High temperature				2								
Low temperature						2						
Drop												
High temperature				2								
Normal temperature					1							
Low temperature												
Acceleration												
High temperature												2
Low temperature												1
Accelerated age										2		
Altitude firing												
High temperature	1			1	1	1		1		1		
Normal temperature		1							1		2	
Low temperature			1	1		1	1			1		
Temperature gradient												
High to low									1			
Low to high									1			

[redacted] [redacted]

Table 5-10. Program for Test Articles, Hot Wire Igniter Cartridge

Test	Test Sequence and No. of Units									
	20	20	20	10	10	20	4	4	6	4
Acceptance	1	1	1	1	1	1	1	1	1	1
No fire current	2	2	2	2	2	2	2	2	2	2
Static sensitivity	3	3	3	3	3	3	3	3	3	3
Dielectric strength	4	4	4	4	4	4	4	4	4	4
Humidity	5									
Eight-foot drop				5						
Salt spray						5				
Vibration and temperature	6	5	5							
High temperature		6								
Low temperature			6							
Fire - room temperature				6	6		6	6	6	6
Fire - high temperature	7	7								
Fire - low temperature			7			6				
Vibration (off limit)					5					
Jolt							5			
Jumble								5		
Forty-foot drop									5	
High temperature tolerance										5

5.4 EARTH LANDING SYSTEM

5.4.1 Scope

The purpose of the ground qualification test program is to qualify the earth landing system by subjecting components to mission environments; i.e., humidity, high and low temperature vacuum, vibration, acceleration, and shock. These ground tests are in addition to the drop test program.

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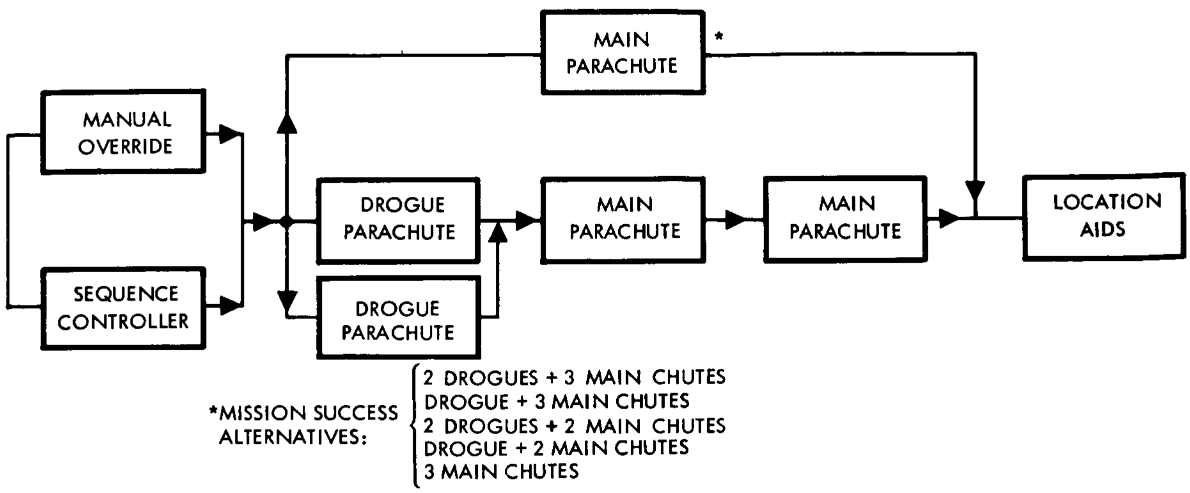

Figure 5-14. ELS Reliability Assessment Model

5.4.2 Reliability Assessment Model

Figure 5-14 depicts the reliability assessment model for the ELS. The model will aid in establishing test continuity and in programming the collection of data for the assessment of reliability by illustrating the interactions of the system components and subassemblies.

5.4.3 Criticality and Hardware Requirements

Table 5-11 designates the criticality class, as defined in subsection 2.4, for the components of the ELS. The table enumerates the hardware requirements for a minimum qualification program as established in subsection 2.5. The distribution of the hardware through the subsystem qualification test phases, as described in subsection 2.3, is shown in the respective qualification test schedule. The location of the components in the spacecraft is indicated by spacecraft zone numbers, as defined in subsection 4.2.

5.4.4 Qualification Test Schedule

5.4.4.1 Drogue and Main Parachute Disconnects

Figure 5-15 displays the qualification test schedule for the ELS parachute disconnects and defines the test utilization of the hardware enumerated in Table 5-11. The specific test levels to be applied in qualification testing are found in the tables of environmental test criteria in subsection 4.2 and

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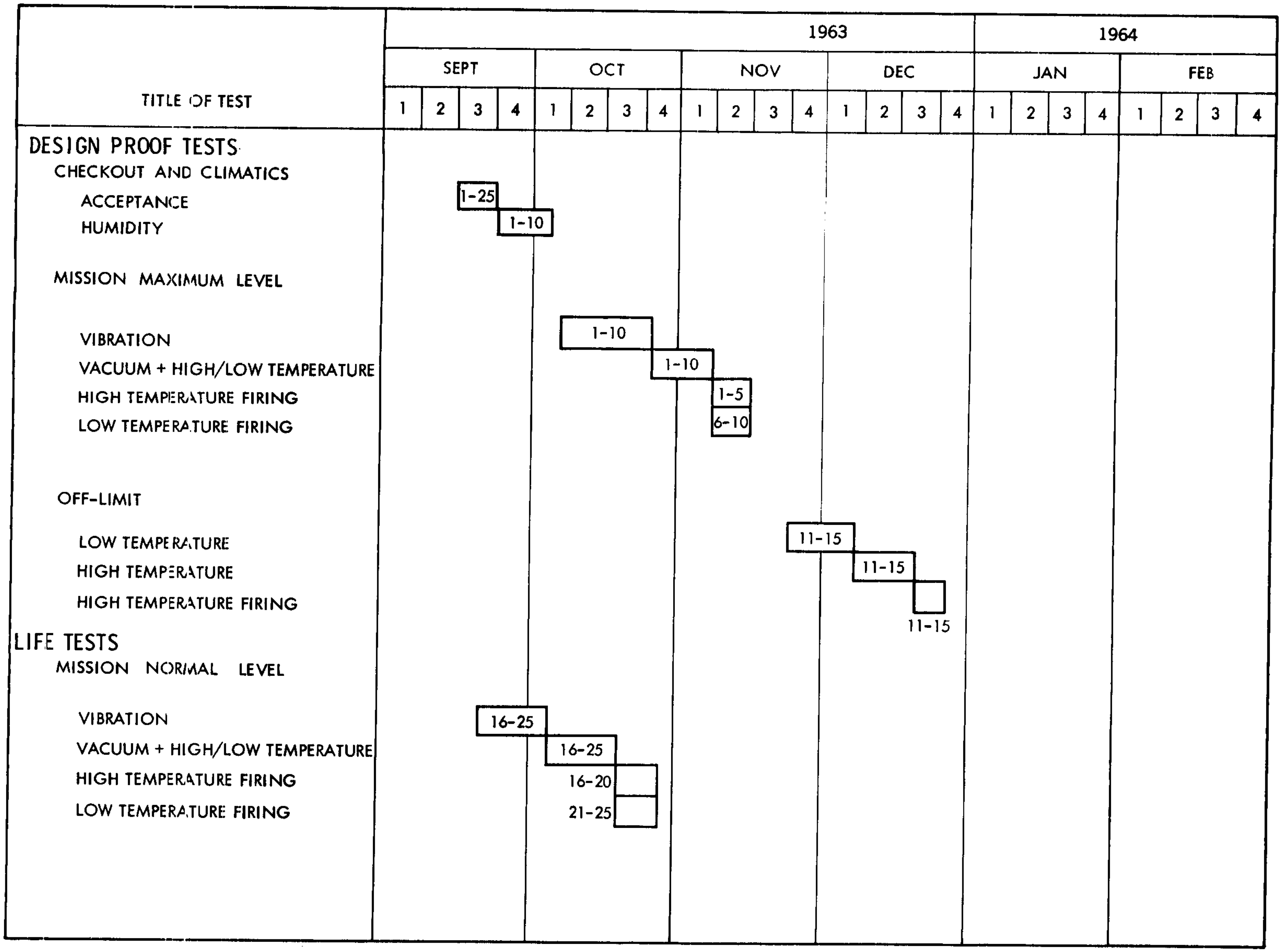

Figure 5-15. ELS Main Parachute Disconnects Qualification Test Schedule

[redacted]

Table 5-11. ELS Parameters for Qualification Testing

	S/C Zone	Criticality	Required Hardware
Main chute disconnect	1	3	25
Drogue chute disconnect	1	3	25
Drogue chute mortar	1	3	4
Pilot chute mortar	1	3	4
Sequence controller	1	3	4
Parachute fabric sample	1	3	25 of each specimen
Main cluster harness assy	1	3	44
Reefing line cutter	1	3	50

are a function of the zone location of the equipment in the spacecraft. These zone numbers are given in Table 5-11.

5.4.4.2 Drogue and Pilot Mortars

Figure 5-16 displays the qualification test schedule for the ELS mortars and defines the test utilization of the hardware enumerated in Table 5-10. The specific test levels to be applied in qualification testing are found in the tables of environmental test criteria in subsection 4.2 and are a function of the zone location of the equipment in the spacecraft. These zone numbers are given in Table 5-11.

5.4.4.3 Sequence Controller

Figure 5-17 displays the qualification test schedule for the ELS sequence controller and defines the test utilization of the hardware enumerated in Table 5-11. The specific test levels to be applied in qualification testing are found in the tables of environmental test criteria in subsection 4.2 and are a function of the zone location of the equipment in the spacecraft. These zone numbers are given in Table 5-11.

5.4.4.4 Parachute Fabric Samples

Figure 5-18, 5-19, and 5-20 display the qualification test schedule for the ESL parachute fabric samples and defines the test utilization of the hardware enumerated in Table 5-11. The specific test levels to be applied in qualification testing are found in the tables of environmental test criteria in

TITLE OF TEST	1963												1964											
	OCT				NOV				DEC				JAN				FEB				MAR			
	1	2	3	4	1	2	3	4	1	2	3	4	1	2	3	4	1	2	3	4	1	2	3	4
DESIGN PROOF TESTS																								
CHECKOUT AND CLIMATICS																								
ACCEPTANCE	1	3-4			2																			
HUMIDITY		1																						
MISSION MAXIMUM LEVEL																								
VIBRATION			1																					
VACUUM + HIGH/LOW TEMPERATURE				1																				
LOW TEMPERATURE FIRING					1																			
HIGH TEMPERATURE FIRING						1																		
SHOCK							1																	
OFF-LIMIT																								
LOW TEMPERATURE						2																		
HIGH TEMPERATURE							2																	
HIGH TEMPERATURE FIRING								2																
SHOCK									2															
LIFE TESTS																								
MISSION NORMAL LEVEL																								
VIBRATION				3-4																				
VACUUM + HIGH/LOW TEMPERATURE					3-4																			
HIGH TEMPERATURE FIRING						3-4																		
LOW TEMPERATURE FIRING							3-4																	
SHOCK								3-4																

Figure 5-16. ELS Drogue and Pilot Mortars Qualification Test Schedule

[redacted]

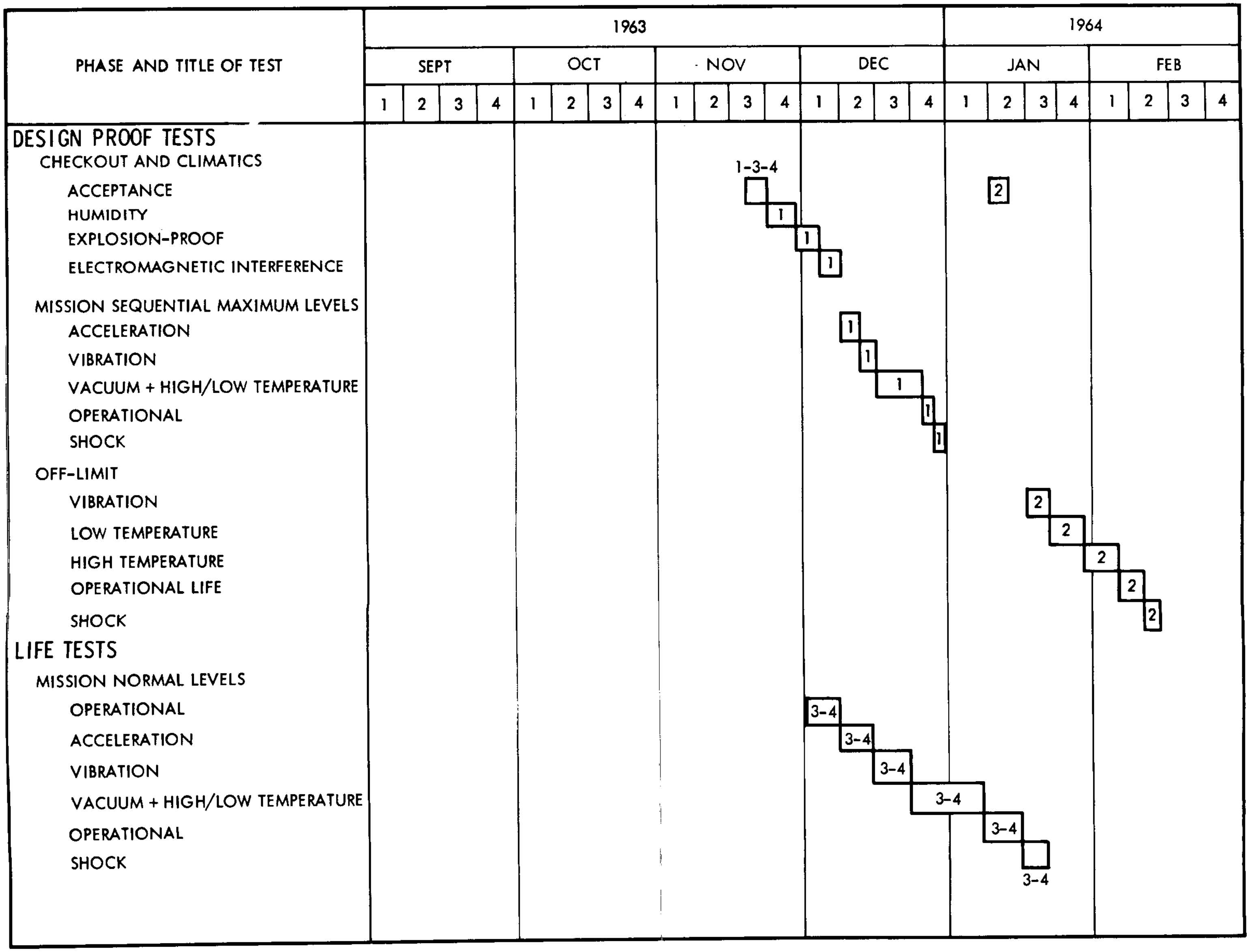

Figure 5-17. ELS Sequence Controller Qualification Test Schedule

TITLE OF TEST	1963 SEPT 3	SEPT 4	OCT 1	OCT 2	OCT 3	OCT 4	NOV 1	NOV 2	NOV 3	NOV 4	DEC 1	DEC 2	DEC 3	DEC 4	1964 JAN 1	JAN 2	JAN 3	JAN 4	FEB 1	FEB 2	FEB 3	FEB 4	MAR 1	MAR 2
DESIGN PROOF TESTS																								
CHECKOUT AND CLIMATICS																								
ACCEPTANCE	1–5				6–10				11–25															
HUMIDITY		1–5																						
STORAGE, HIGH TEMPERATURE			1–5																					
MISSION MAXIMUM LEVEL																								
VACUUM + HIGH/LOW TEMPERATURE				1–5																				
PULL TEST					1–5																			
OFF-LIMIT																								
HIGH TEMPERATURE 175 F																								
HIGH TEMPERATURE 200 F																								
HIGH TEMPERATURE 225 F										11–25														
HIGH TEMPERATURE 250 F																								
HIGH TEMPERATURE 300 F																								
PULL TEST											11–25													
LIFE TESTS																								
MISSION NORMAL LEVEL																								
VACUUM + HIGH/LOW TEMPERATURE						11–25																		
PULL TEST							11–25																	

Figure 5-18. ELS Parachute Fabric Samples Qualification Test Schedule

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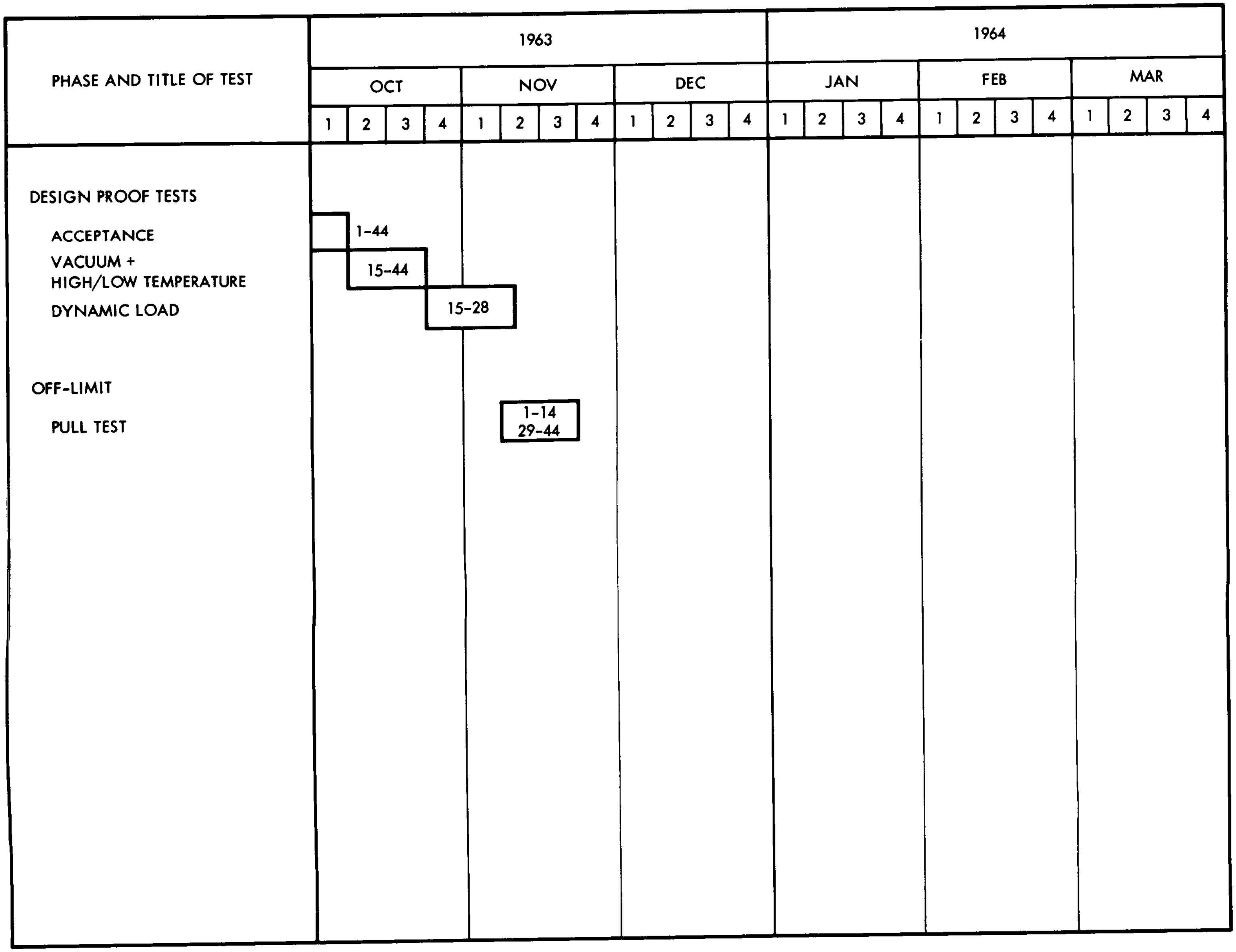

Figure 5-19. ELS Main Cluster Harness Assembly Qualification Test Schedule

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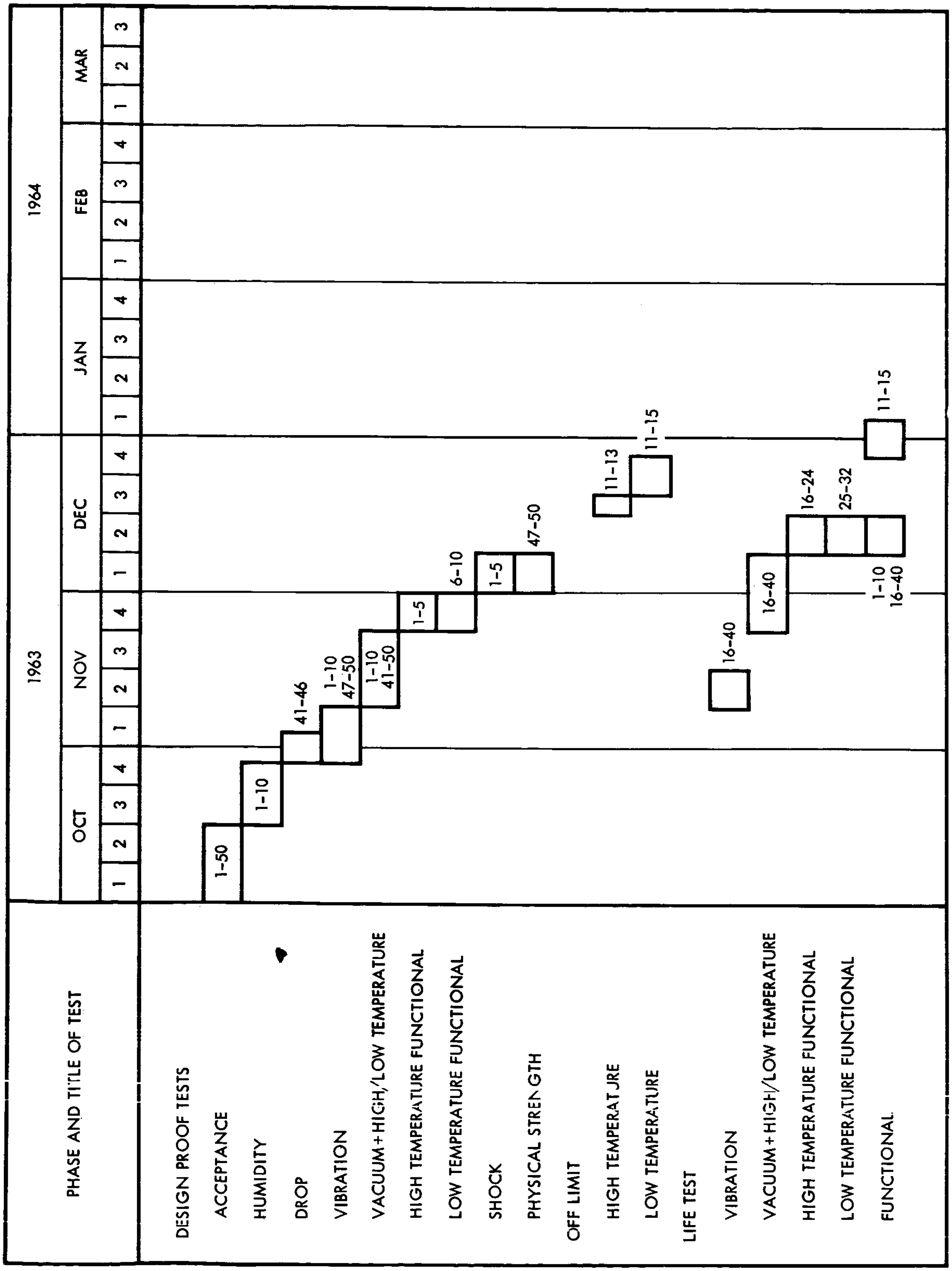

Figure 5-20. ELS Reefing Line Cutter Qualification Test Schedule

[redacted]

subsection 4.2 and are a function of the zone location of the equipment in the spacecraft. These zone numbers are given in Table 5-11.

5.5 ENVIRONMENTAL CONTROL SYSTEM AND CRYOGENIC SUBSYSTEM

5.5.1 Scope

5.5.1.1 Environmental Control System

Complexity of the subsystem and its circuits with many components not in a common housing make qualification as a system difficult. For this reason, the subcontractor qualification program has been designed at the component and package level. The subcontractor will deliver components and packages to S&ID where they will be mounted in command module structures and tested as a system in boilerplate and airframe tests.

5.5.1.2 Cryogenic Subsystem

Laboratory qualification of the cryogenic storage system and reliability assessment from data obtained will be performed by the subcontractor.

5.5.2 Reliability Assessment Model

5.5.2.1 Environmental Control System

Figure 5-21 depicts the reliability assessment model for the ECS. The model will aid in establishing test continuity and in programming the collection of data for the assessment of reliability by illustrating the interfaces of the system circuits and components.

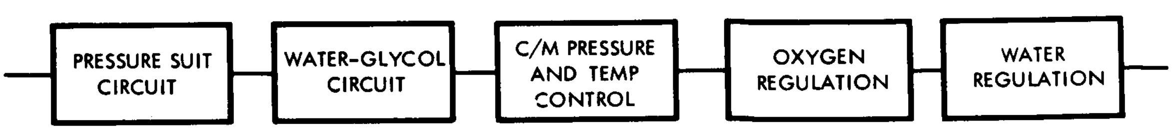

Figure 5-21. ECS Test Oriented Reliability Logic Diagram

5.5.2.2 Cryogenic Subsystem

Figure 5-22 depicts the reliability assessment model for the cryogenic subsystem. The model will aid in establishing test continuity and in

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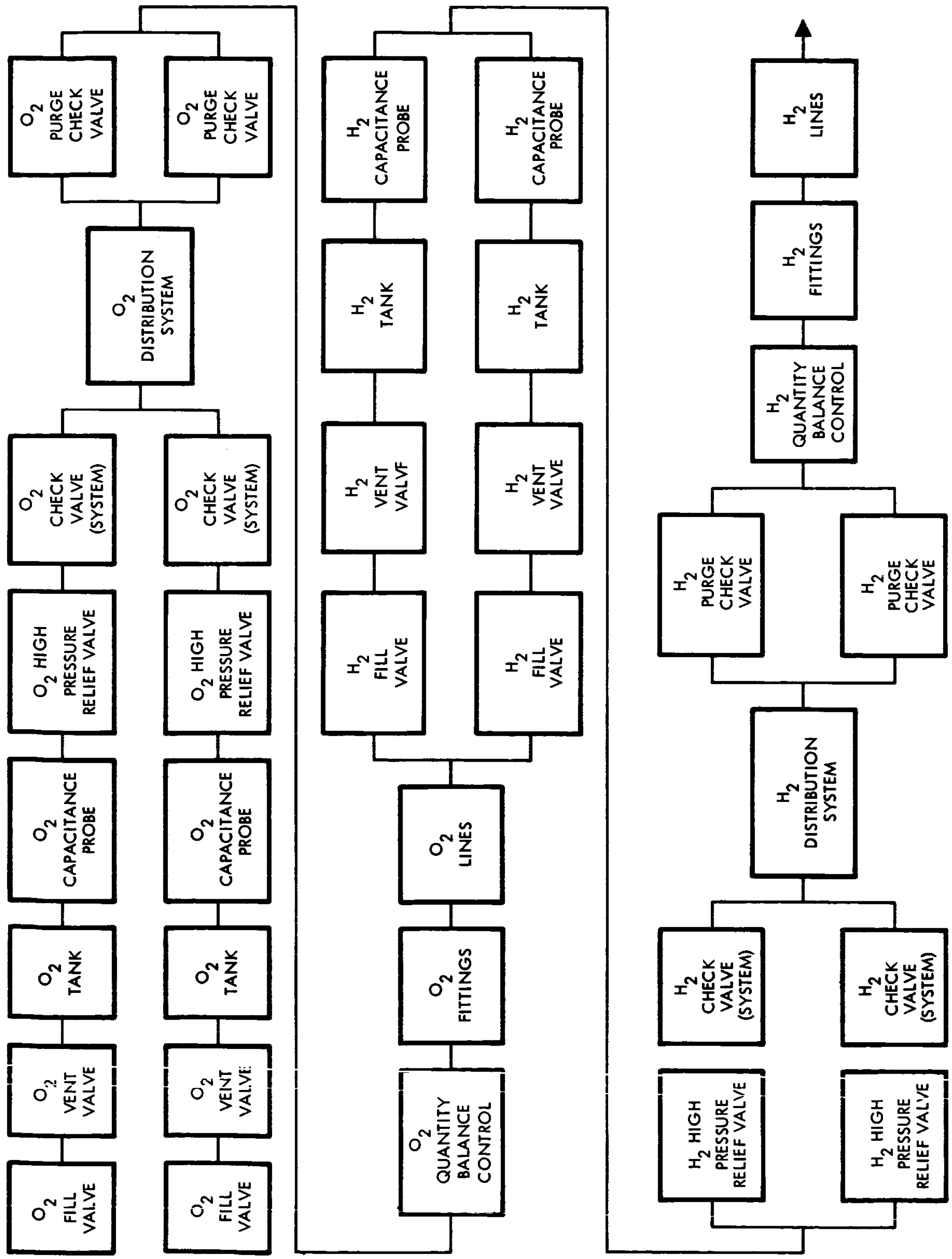

Figure 5-22. Cryogenic Reliability Assessment Model

[redacted]

programming the collection of data for the assessment of reliability by illustrating the interaction of the system components.

5.5.3 Criticality and Hardware Requirements

5.5.3.1 Environmental Control System

Table 5-12 designates the criticality class, as defined in subsection 2.4, for the components of the ECS. The table enumerates the hardware requirements for a minimum qualification program as established in subsection 2.5. The location of the components in the spacecraft is indicated by spacecraft zone numbers as defined in subsection 4.2.

Table 5-12. ECS Parameters for Qualification Testing

Component	S/C Zone	Criticality	Required Hardware
Pressure suit	1	1	3
Water-glycol	1	1	3
C/M pressure and temperature control	1	1	3
Oxygen supply system	1	1	3
Water supply system	1	1	3
Transducers	1	1	3
Cold plates	2	1	3
Space radiators	1 and 3	1	3

5.5.3.2 Cryogenic Subsystem

Table 5-13 designates the criticality class, as defined in subsection 2.4, for the components of the cryogenic subsystem. The table enumerates the hardware requirements for a minimum qualification program as established in subsection 2.5. The location of the components in the spacecraft is indicated by spacecraft zone numbers as defined in subsection 4.2.

Table 5-13. Cryogenic Subsystem Parameters for Qualification Testing

Component	S/C Zone	Criticality	Required Hardware
Oxygen subsystem	1 and 3	3	4 (less one tank each)
Hydrogen subsystem	1 and 3	3	4 (less one tank each)

[redacted] [redacted]

5.5.4 Qualification Test Schedule

5.5.4.1 Environmental Control System

Figure 5-23 displays the qualification test schedule for the ECS and defines the test utilization of the hardware enumerated in Table 5-12. The specific test levels to be applied in qualification testing are found in the tables of environmental test criteria in subsection 4.2 and are a function of the zone location of the equipment in the spacecraft. These zone numbers are given in Table 5-12.

5.5.4.2 Cryogenic Subsystem

Figures 5-24 and 5-25 display the qualification test schedule for the cryogenic subsystem and define the test utilization of the hardware enumerated in Table 5-13. The specific test levels to be applied in qualification testing are found in the tables of environmental test criteria in subsection 4.2 and are a function of the zone location of the equipment in the spacecraft. These zone numbers are given in Table 5-13.

5.6 ELECTRICAL POWER SYSTEM AND FUEL CELLS

5.6.1 Scope

5.6.1.1 Electrical Power System

The qualification program for the EPS has been designed to qualify the system for Apollo spacecraft use while determining optimum qualification test costs. The tests will verify design integrity and will provide S&ID with data to be integrated with other ground and flight test data for reliability assessment purposes. The components of the EPS are to be supplied to S&ID by different suppliers and subcontractors. Each supplier or subcontractor will perform the test, report and analyze failures, propose corrective action, and furnish S&ID with data to be used for reliability assessment. S&ID will monitor the tests and assist in failure analysis when necessary. S&ID will approve corrective actions, approve qualification results, and assess achieved reliability based on data supplied by the subcontractor or supplier. The qualification sample sizes were selected as the minimum required to accomplish engineering objectives. The number and type of tests will vary for the different components of the EPS depending on its use, operational characteristics, and location within the spacecraft.

5.6.1.2 Fuel Cells

Although its basic theory of operation has been known for decades, the fuel cell is a recent addition to practical power supplies. For this reason,

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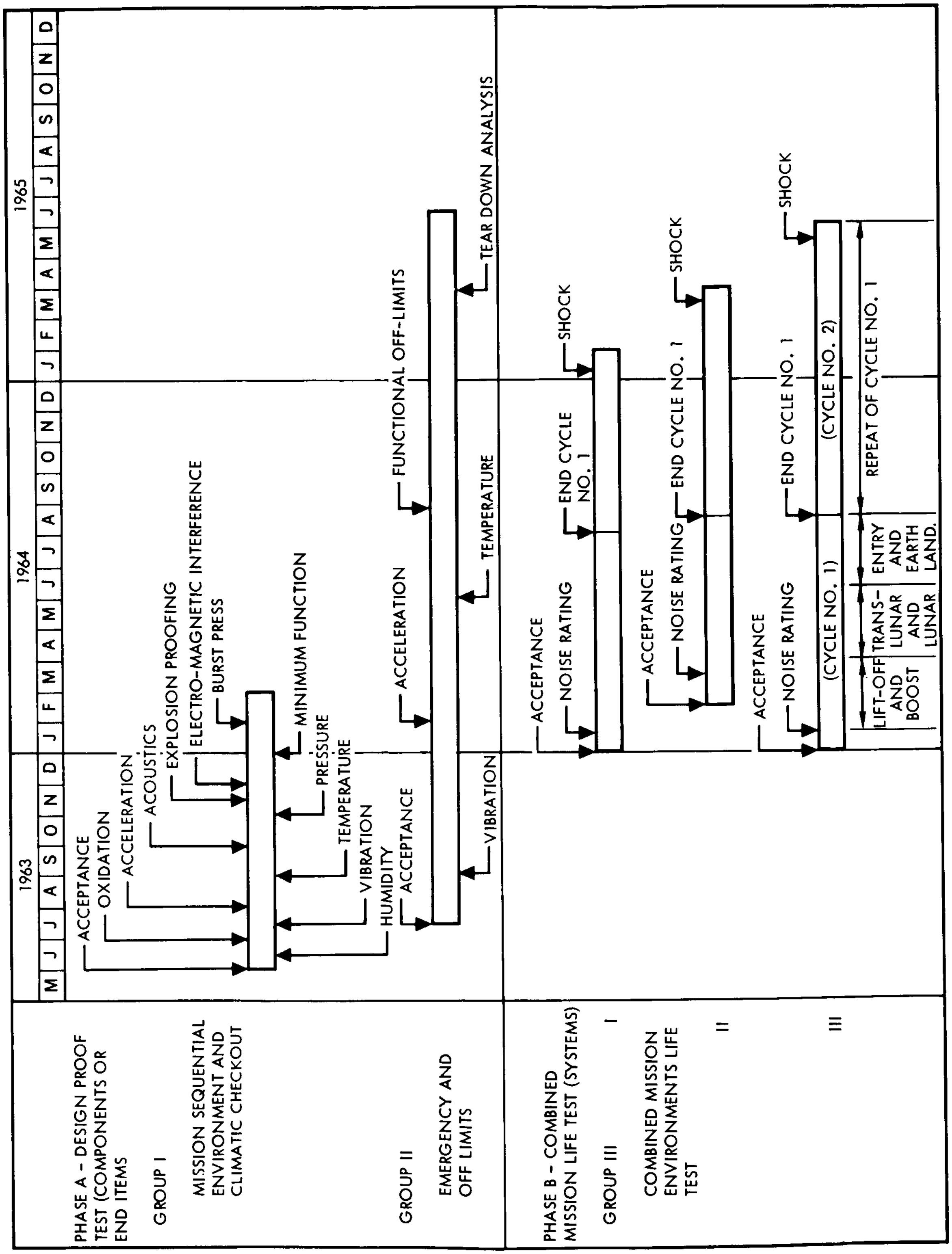

Figure 5-23. ECS Qualification Test Schedule

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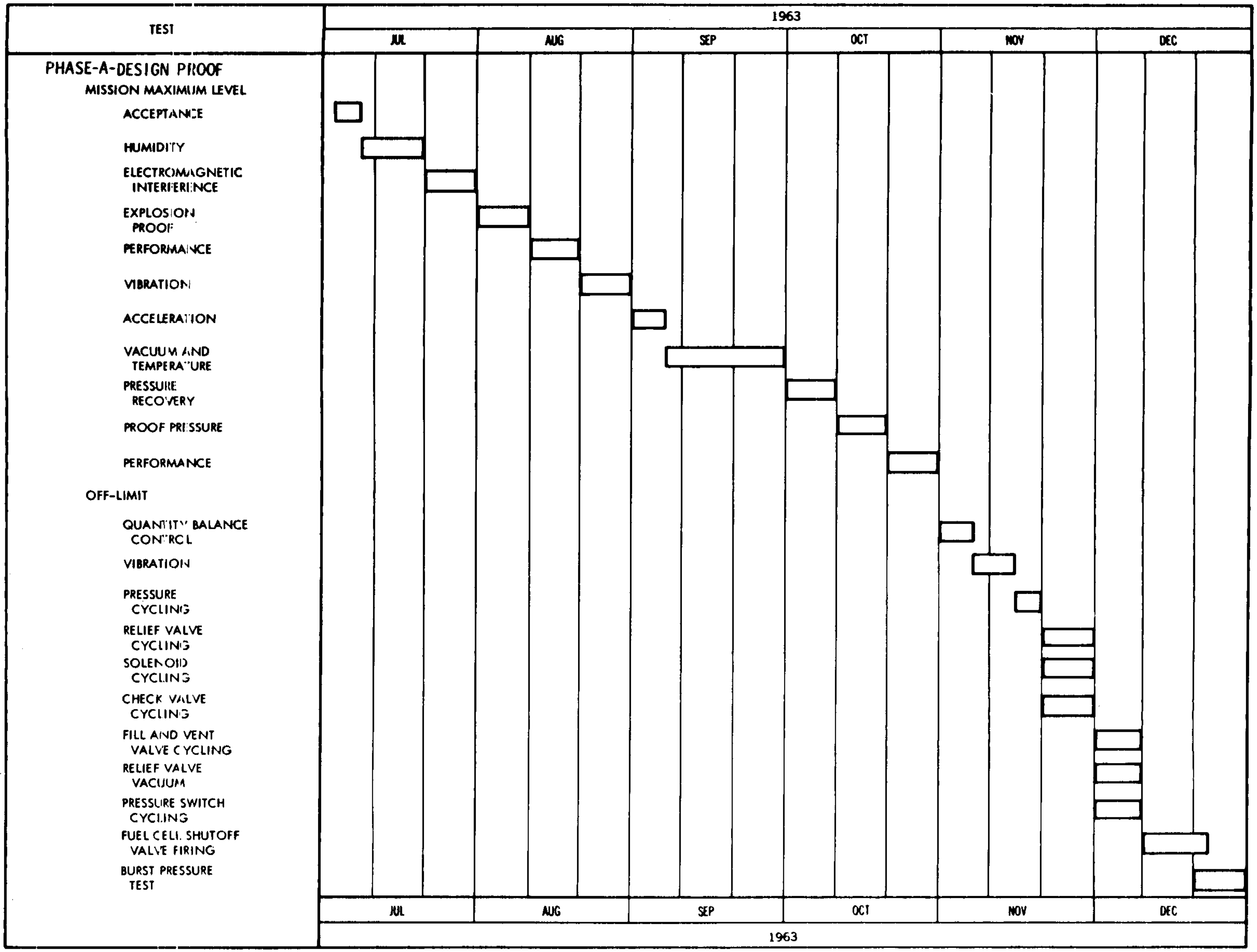

Figure 5-24. Cryogenic Subsystem Qualification Test Schedule, Phase A

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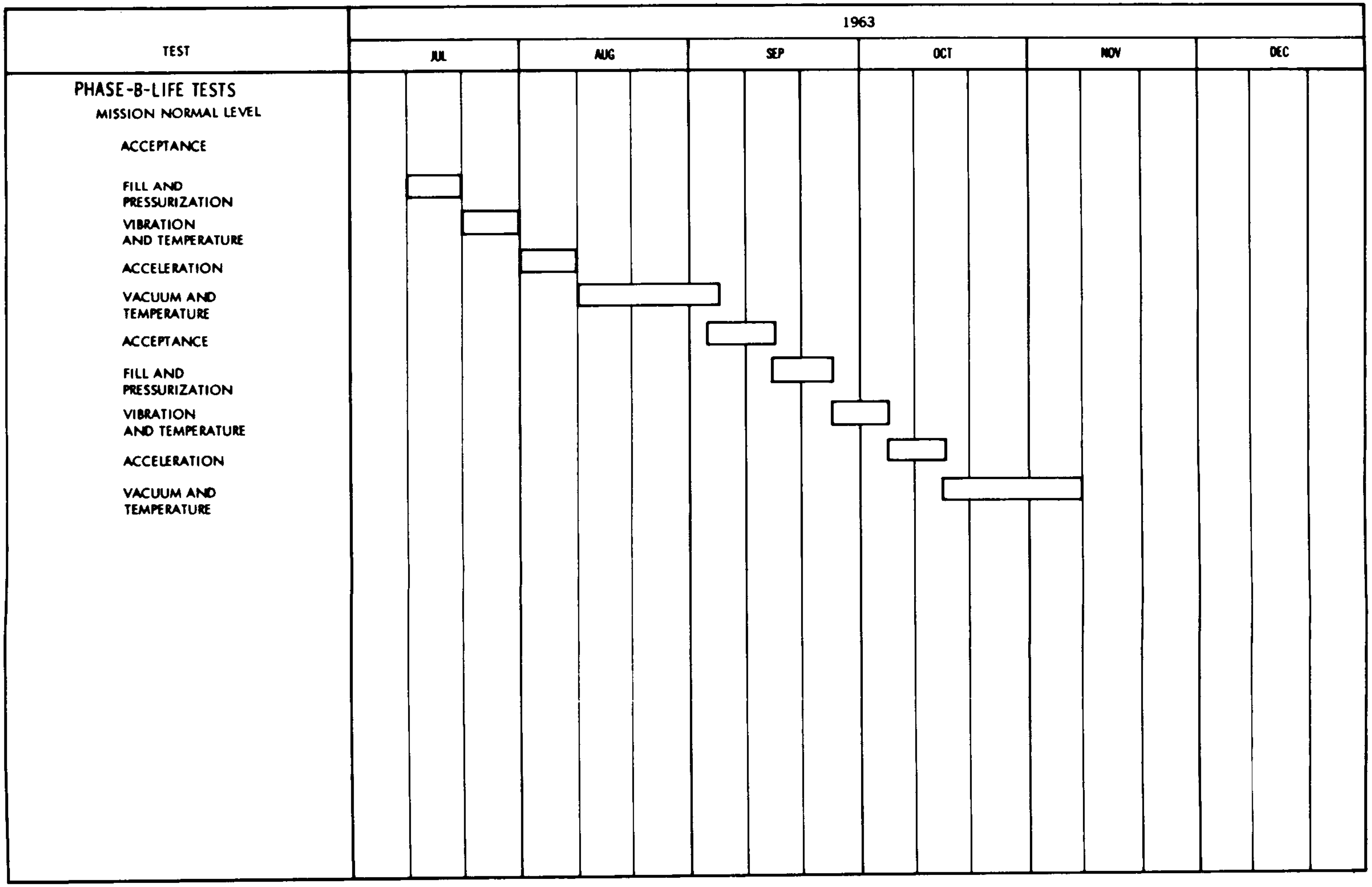

Figure 5-25. Cryogenic Subsystem Qualification Test Schedule, Phase B

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the manufacturing "know-how" is new. The operating life (without replacement of parts) is still relatively short. The Apollo fuel cell power plant, as presently being developed, will have an operating life of 400 hours, which is 20 percent above the lunar mission time. The qualification test program has been carefully planned to verify that the fuel cell design will meet this operating life requirement. Because of the importance of the fuel cell to mission success, an active redundancy is used in the system (i.e., three fuel cells are used in parallel when any two will supply sufficient power).

Fuel cell qualification testing will be conducted by the subcontractor with surveillance by S&ID to insure conformance. Integrated system testing will be accomplished by boilerplate and airframe tests at S&ID.

5.6.2 Reliability Assessment Model

5.6.2.1 Electrical Power System

Figure 5-26 depicts the reliability assessment model for the EPS. The model will aid in establishing test continuity and in programming the collection of data for the assessment of reliability by illustrating the functional and reliability interrelationships of the system components.

5.6.2.2 Fuel Cells

Figure 5-27 illustrates the reliability interrelationship of the fuel cells.

5.6.3 Criticality and Hardware Requirements

5.6.3.1 Electrical Power System

Table 5-14 designates the criticality class, as defined in subsection 2.4, for the components of the EPS less the fuel cells. The table enumerates the hardware requirements for a minimum qualification program as established in subsection 2.5. The distribution of the hardware through the subsystem qualification test phases, as described in subsection 2.3, is shown in the qualification test schedule (Figure 5-28). The location of the components in the spacecraft is indicated in the table by spacecraft zone numbers as defined in subsection 4.2.

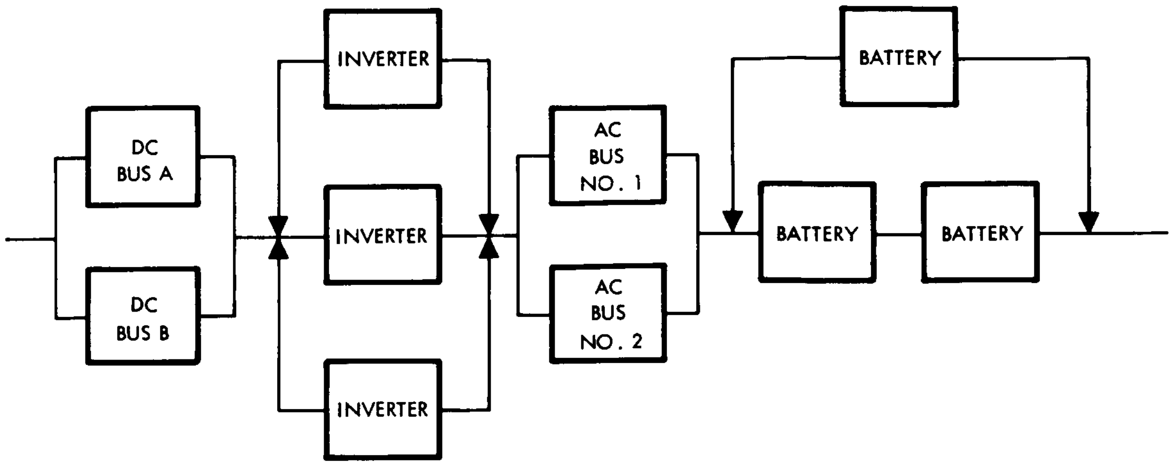

Figure 5-26. EPS Reliability Assessment Model (Less Fuel Cell)

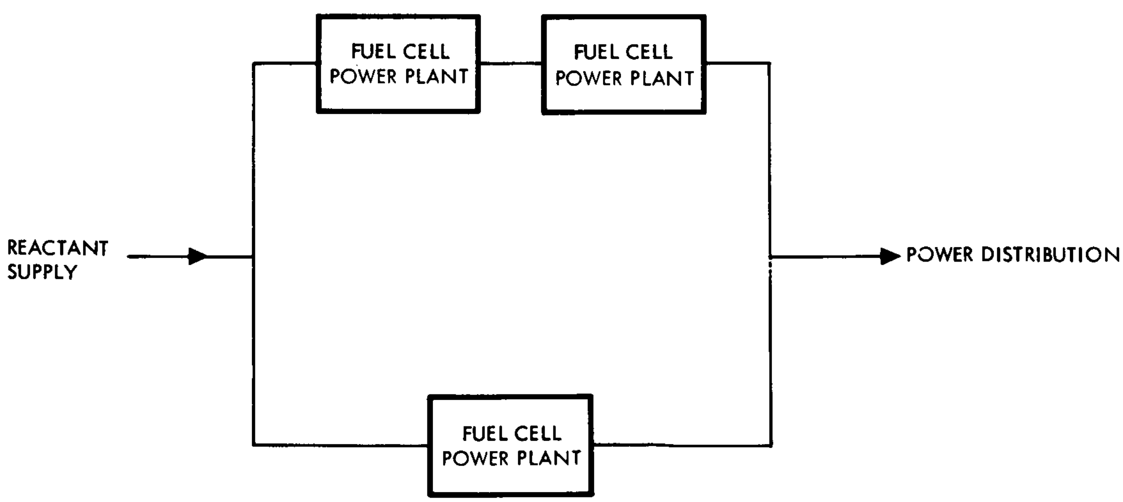

Figure 5-27. Fuel Cell Reliability Assessment Model

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Table 5-14. EPS Parameters for Qualification Testing

	S/C Zone	Criticality	Required Hardware
Static inverter	2	3	4
Battery	2	3	10
Battery charger	2	3	4
Circuit breakers	1 and 2	3	6 of each type 3 types
Relay general purpose	1 and 2	3	36
Connector, umbilical C/M to LES	1 external	3	4
Connector, general purpose electrical system	1, 2, 3	3	3 class F (each size) 3 class H (each size) with 3 class E plugs
Electrical feed through bulkhead C/M	2	3	2 of each type 2 types
Power transfer switch	2	3	6 of each type 3 types
Motor operated power switch and overcurrent relay	3	3	5
Floodlight fixtures	2	3	14 total 3 types
Ribbon cables C/M to S/M	1 external	3	6
Cable, electrical flat, 600 volts	1 external	3	6
Sequencer	2	3	
Relay - A			2
Relay - B			2
Solid state - C			2
Solid state - D			4
RCS transfer switch	1	3	4 of each class
Motor-operated DPST			
Transfer switch			4
Electrical connector set	1 internal	3	6
Aft and forward bulkhead connector	1 external	3	5
Rectangular electrical connectors	1 external	3	3

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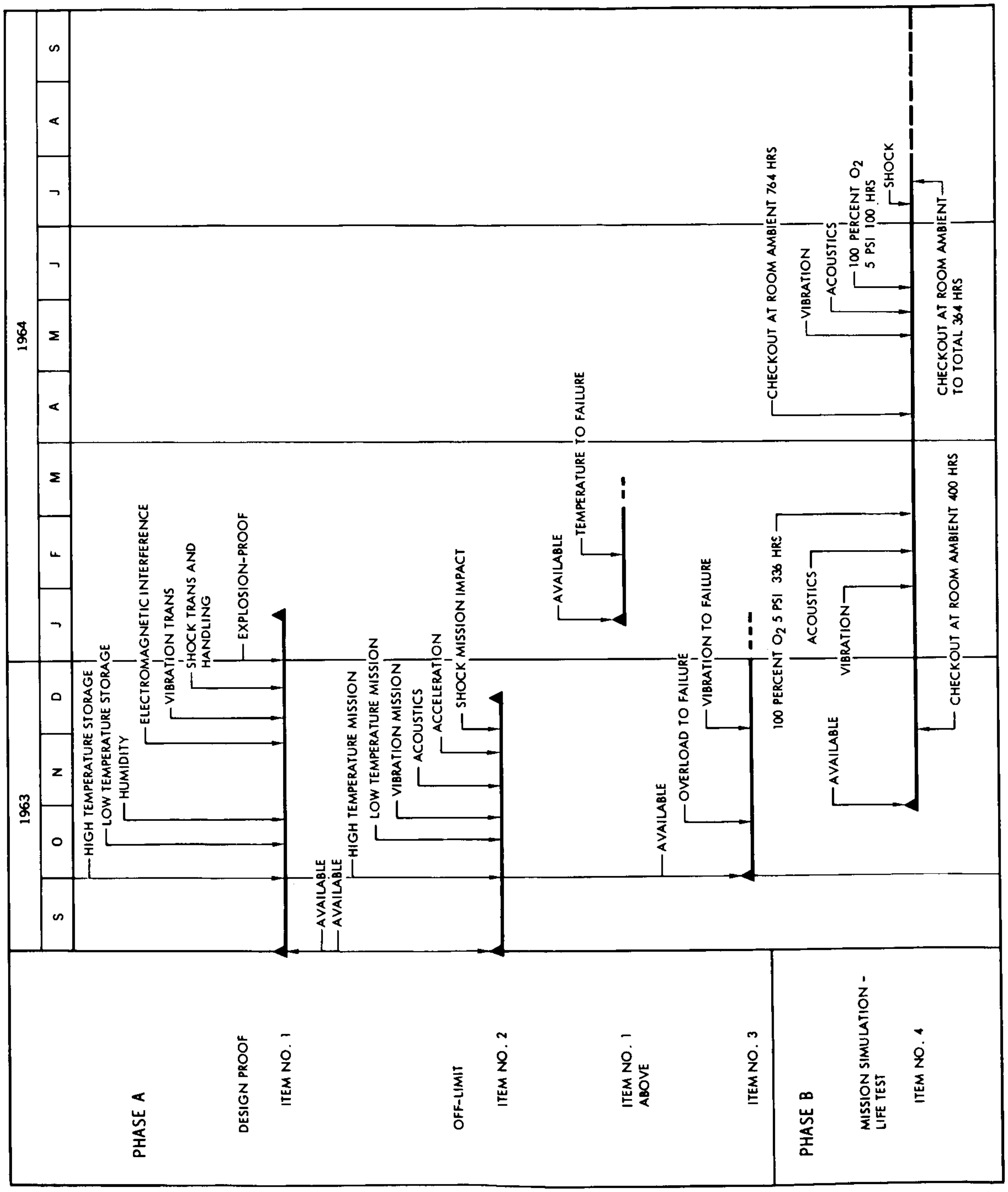

Figure 5-28. EPS Qualification Test Schedule

[redacted]AL

[redacted] [redacted]

5.6.3.2 Fuel Cells

Table 5-15 designates the criticality class, as defined in subsection 2.4, for the components of the fuel cells. The table enumerates the hardware requirements for a minimum qualification program, as established in subsection 2.5. The distribution of the hardware through the subsystem qualification test phases, as described in subsection 2.3, is shown in the qualification test schedule (Figure 5-29). The location of the components in the spacecraft is indicated in the table by spacecraft zone numbers, as defined in subsection 4.2.

Table 5-15. Fuel Cell Parameters for Qualification Testing

Component	S/C Zone	Criticality	Required Hardware
Fuel Cell	3	3	4*
*One refurbishment is required on Unit 2.			

5.6.4 <u>Qualification Test Schedule</u>

5.6.4.1 Electrical Power System

Figure 5-28 displays the qualification test schedule for the EPS and defines the test utilization of the hardware enumerated in Table 5-14. The specific test levels to be applied in qualification testing are found in the tables of environmental test criteria in subsection 4.2 and are a function of the zone location of the equipment in the spacecraft. These zone numbers are given in Table 5-14.

5.6.4.2 Fuel Cells

Figure 5-29 displays the qualification test schedule for the fuel cells and defines the test utilization of the hardware enumerated in Table 5-15. The specific test levels to be applied in qualification testing are found in the tables of environmental test criteria in subsection 4.2 and are a function of the zone location of the equipment in the spacecraft. These zone numbers are given in Table 5-15.

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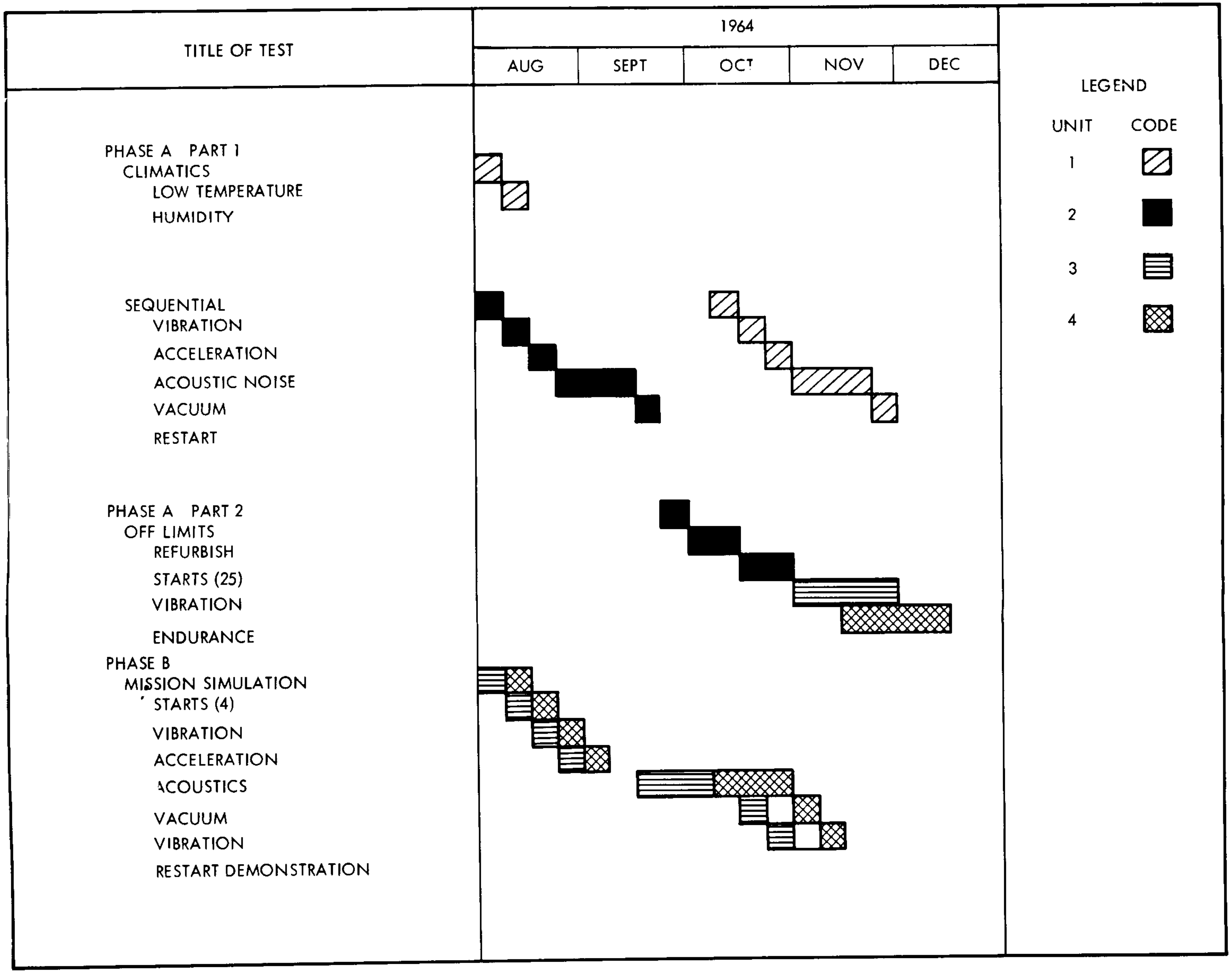

Figure 5-29. EPS Fuel Cells Qualification Test Schedule

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5.7 GUIDANCE AND NAVIGATION SYSTEM

5.7.1 Scope

Efforts concerning the qualification testing of the guidance and navigation components, subsystems, and systems are specifically divided between the associate contractor and S&ID. The test programs conducted by the associate contractor and S&ID will be integrated to avoid duplication of effort and to provide maximum engineering confidence in the success of the Apollo mission.

5.7.2 Reliability Assessment Model

Figure 5-30 depicts the reliability assessment model for the G&NS. The model illustrates the reliability and functional interrelationships among the system components.

5.7.3 Criticality and Hardware Requirements

Table 5-16 designates the criticality class, as defined in subsection 2.4, for the components of the G&NS. The location of the components in the spacecraft is indicated by spacecraft zone numbers as defined in subsection 4.2.

Table 5-16. G&NS Parameters for Qualification Testing

Components	S/C Zone	Criticality	Required* Hardware
IMU	2	2	
PSA	2	2	
CDU	2	2	
D&C	2	2	
AGC	2	2	
OPTICS	2	2	
MDV	2	2	
NAV BASE	2	2	

*To be determined by associate contractor

5.7.4 Qualification Test Schedule

Figure 5-31 displays the qualification test schedule for the G&NS. The qualification test program is negotiated between MIT and NASA.

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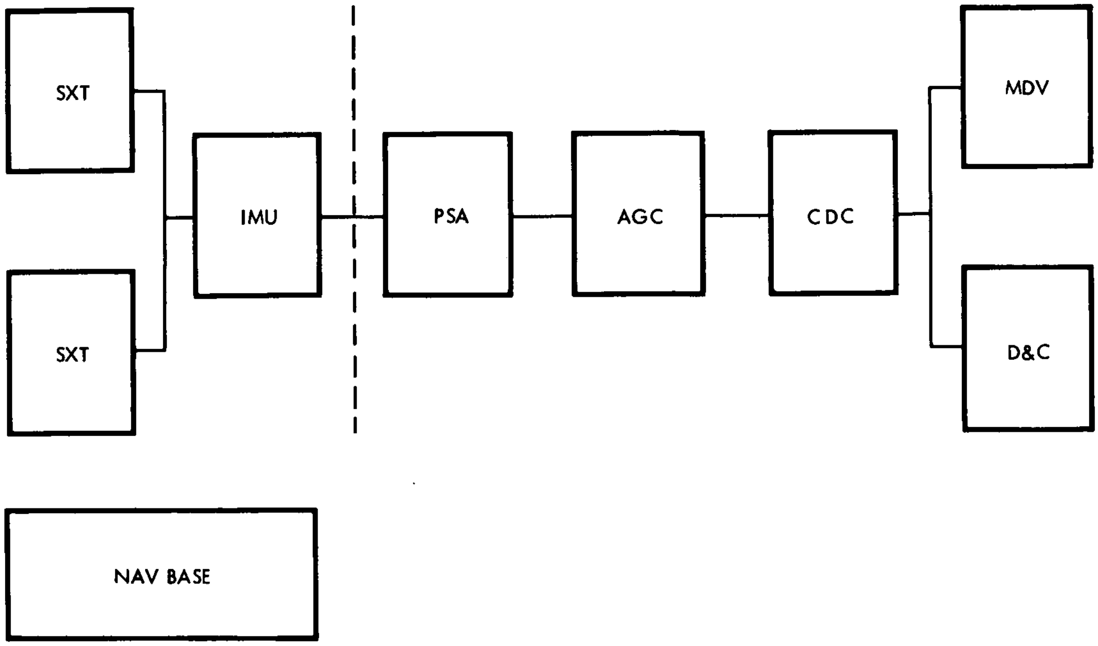

Figure 5-30. Guidance and Navigation Reliability Assessment Model

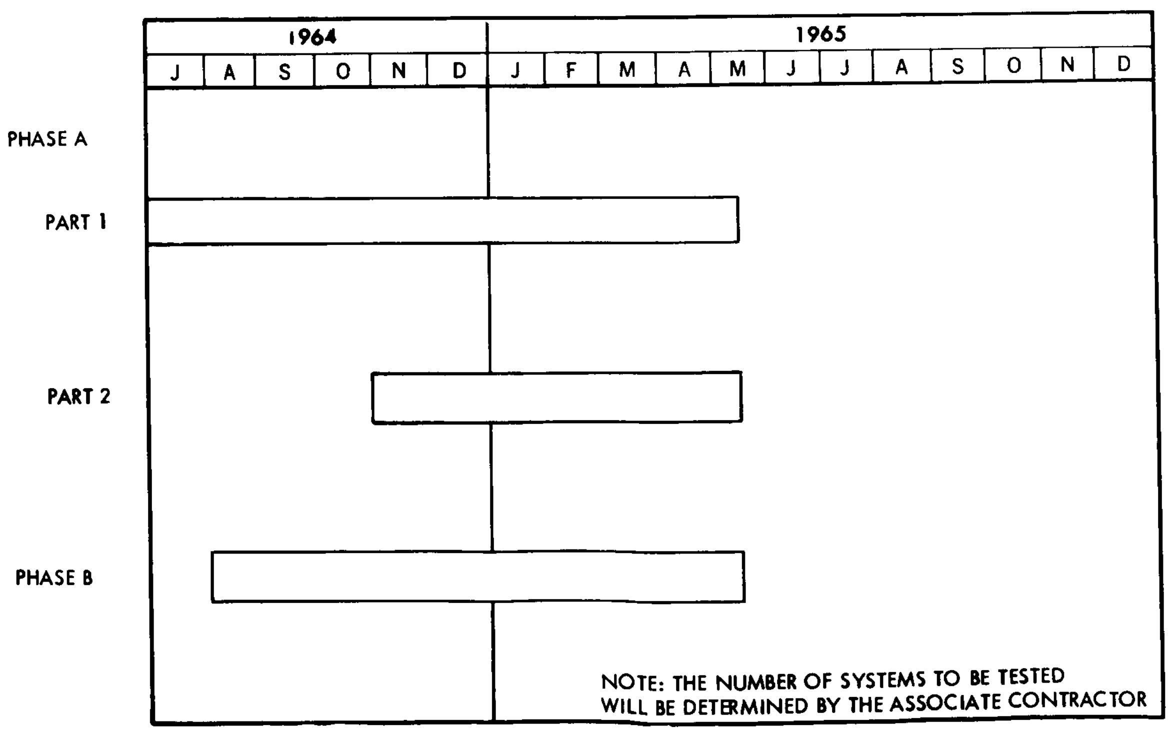

Figure 5-31. Guidance and Navigation Qualification Test Schedule

5.8 STABILIZATION AND CONTROL SYSTEM

5.8.1 Scope

The SCS qualification program has been designed to qualify the stabilization and control subsystem for Apollo use while optimizing qualification test costs. The tests will accomplish all engineering objectives and will provide S&ID with data to be integrated with all other ground and flight test data for reliability assessment purposes. The tests will be performed by the SCS subcontractor, Minneapolis-Honeywell Regulator Company, at their Aeronautical Division in Minneapolis, Minnesota. Honeywell will perform the tests, report and analyze failures, propose corrective action, and furnish S&ID with data to be used for reliability assessment. S&ID will monitor the tests, assist in failure analysis when necessary, approve corrective actions, approve qualification results and assess achieved reliability based on subcontractor-furnished data. The qualification sample sizes were selected as the minimum required to accomplish program objectives and represent the optimum hardware usage. Time and schedule constraints have necessitated the use of additional packages to bring the total sample size to six complete systems in cases where all systems are undergoing concurrent testing.

In effect, the six stabilization and control systems and the two equivalent systems, which are refurbished from the six, serve as eight systems available to accelerate the qualification program. Despite this stratagem, qualification of the SCS will not be completed before the first manned flight, reference paragraph 3.3.8 and Figure 3-9.

5.8.2 Reliability Assessment Model

Figure 5-32 depicts the reliability assessment model for the S&C system. The model will aid in establishing test continuity and in programming the collection of data for the assessment of reliability by illustrating the reliability and functional interrelationships of the system components.

5.8.3 Criticality and Hardware Requirements

Table 5-17 designates the criticality class, as defined in subsection 2.4, for the components of the SCS. The table enumerates the hardware requirements for a minimum qualification program as established in subsection 2.5. The distribution of the hardware through the subsystem qualification test phases as described in subsection 2.3, is shown. The location of the components in the spacecraft is indicated by spacecraft zone numbers as defined in subsection 4.2.

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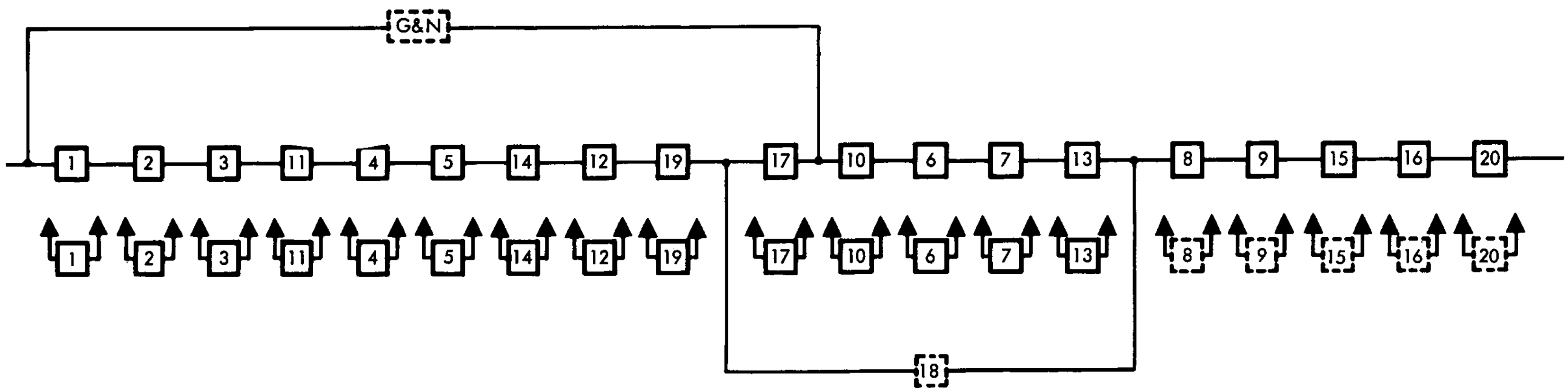


FUNCTIONAL BACK-UP PROVIDED BY OTHER SPACECRAFT SYSTEMS AND EMERGENCY MANUAL CONTROLS.

EQUIPMENT KEY

1. ATTITUDE GYROS - (BMAG)
2. ATTITUDE GYRO ELECTRONICS - (ECA-D)
3. ATTITUDE GYRO COUPLING UNIT - (ECA-X)
4. ACCELEROMETER AND ELECTRONICS - (BMAG)
5. INTEGRATOR - (ECA-D)
6. RATE GYROS - (RGP)
7. RATE GYRO ELECTRONICS - (RGP)
8. THRUST VECTOR CONTROL CIRCUITS - (ECA-P, Y)
9. ENGINE THRUST CIRCUITS - (ECA-X)
10. ATTITUDE CONTROL AND REACTION JET CIRCUITS - (ECA-P, Y, R)
11. ATTITUDE INDICATOR AND DEA - (FDAI, ECA-D)
12. ATTITUDE ERROR INDICATOR AND DEA - (FDAI, ECA-D)
13. ANGULAR VELOCITY INDICATOR AND DEA - (FDAI, ECA-D)
14. ΔV INDICATOR AND DEA - (VI, ECA-D)
15. GIMBAL POSITION INDICATOR AND DEA - (GPI, ECA-D)
16. DEA POWER SUPPLY - (ECA-D)
17. MANUAL ROTATIONAL CONTROLS, NORMAL
18. MANUAL ROTATIONAL CONTROLS, EMERGENCY
19. MANUAL TRANSLATIONAL THRUST CONTROLS, NORMAL
20. SCS CONTROL PANEL - (SCS-CP)

Figure 5-32. SCS Reliability Assessment Model

[redacted]

Table 5-17. System Parameters for Qualification Testing

Component	S/C Zone	Criticality	Required Hardware	Qualification Test A-1	Phase A-2	Dist. B
Rate gyro package	2	1	7	3	*2	4
Attitude gyro and accelerometer package	2	2	5	3	2	2
Electronics control assembly						
ECA - Pitch	2	1	6	3	2	3
ECA - Roll	2	1	6	3	2	3
ECA - Yaw	2	1	6	3	2	3
ECA - Auxiliary	2	1	6	3	2	3
ECA - Display	2	2	6	3	2	3
Gimbal position indicator	2	2	6	3	2	3
Three-axis-rotational control	2	2	6	3	2	3
Translation control	2	2	6	3	2	3
V display	2	2	6	3	2	3
Flight director attitude indicator	2	2	6	3	2	3
SCS control panel	2	1	5	3	2	2

*Samples for off limit tests are refurbished units from design proof test (A-1)

5.8.4 Qualification Test Schedule

Figure 5-33 displays the qualification test schedule for SCS and defines the test utilization of the hardware enumerated in Table 5-17. The specific test levels to be applied in qualification testing are found in the tables of environmental test criteria in subsection 4.2 and are a function of the zone location of the equipment in the spacecraft. These zone numbers are given in Table 5-17.

5.9 COMMUNICATIONS AND INSTRUMENTATION SYSTEMS

5.9.1 Scope

5.9.1.1 Television Equipment and Central Timing Unit

Qualification tests on the television equipment and CTU conform to the general requirements of MIL-E-5272 and include the following:

1. Calibration tests
2. Environmental tests
3. Electromagnetic interference
4. Susceptibility tests
5. Mission simulation-life tests

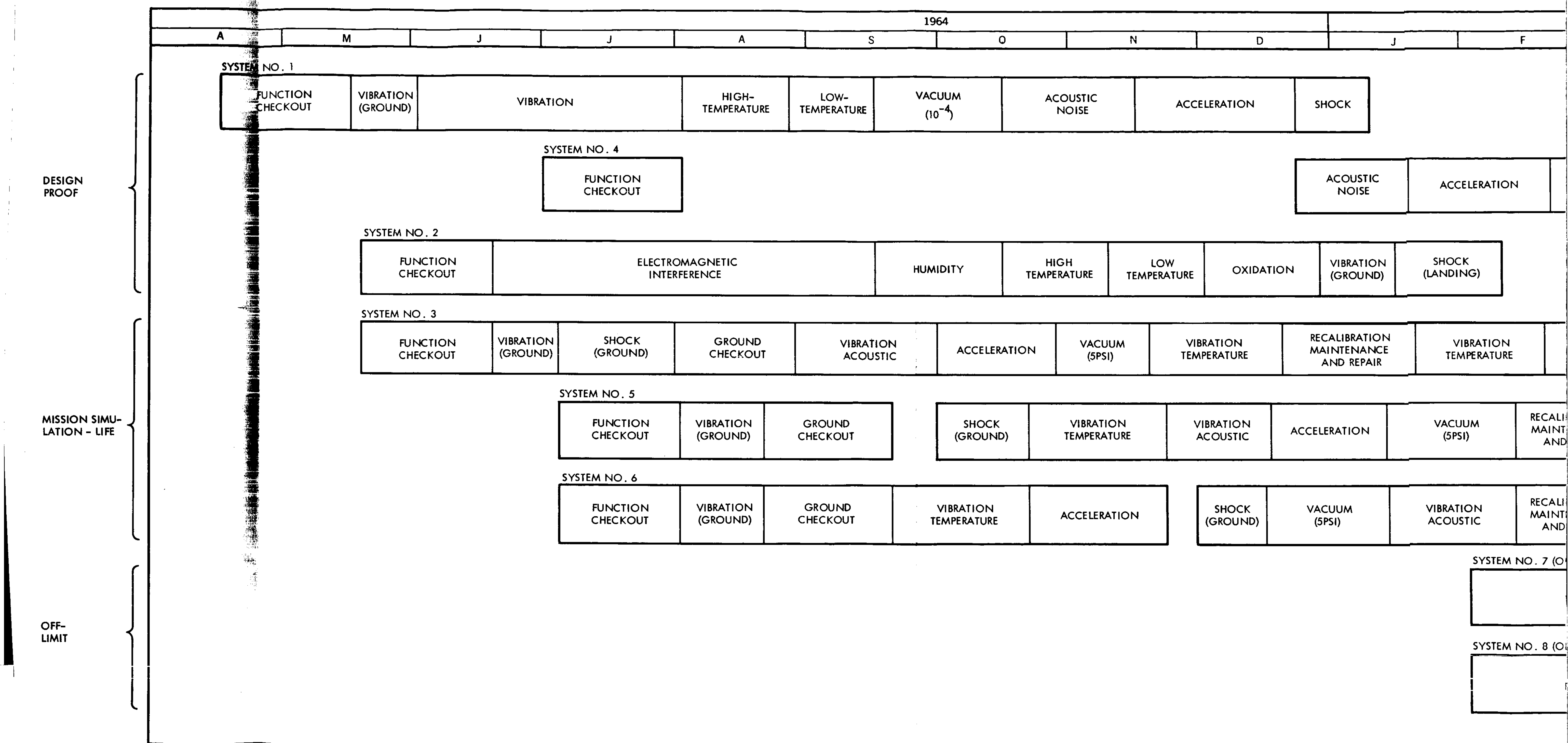
The number of test systems is predicated on the requirement to assess compliance with the procurement specification both in the normal and off-limit modes of operation. The number of qualification test samples of the television equipment and the CTU is based on the criticality index, limited life requirements, service life considerations, and multiple use of test articles.

5.9.1.2 Communications and Data Subsystem

Same as for 5.9.1.1.

5.9.1.3 Antenna Systems

The antenna ground qualification test plan has been designed to accomplish three objectives. First, to test to the maximum levels of the mission environments; second, to test beyond design limits to determine the design margins in critical mission environments; and third, to conduct environmental testing in accordance with the mission sequence. If the antenna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A
M
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A
S
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J
F
DESIGN PROOF
MISSION SIMULATION - LIFE
OFF-LIMIT
SYSTEM NO. 1
FUNCTION CHECKOUT
VIBRATION (GROUND)
VIBRATION
HIGH-TEMPERATURE
LOW-TEMPERATURE
VACUUM (10^{-4})
ACOUSTIC NOISE
ACCELERATION
SHOCK
SYSTEM NO. 4
FUNCTION CHECKOUT
ACOUSTIC NOISE
ACCELERATION
SYSTEM NO. 2
FUNCTION CHECKOUT
ELECTROMAGNETIC INTERFERENCE
HUMIDITY
HIGH TEMPERATURE
LOW TEMPERATURE
OXIDATION
VIBRATION (GROUND)
SHOCK (LANDING)
SYSTEM NO. 3
FUNCTION CHECKOUT
VIBRATION (GROUND)
SHOCK (GROUND)
GROUND CHECKOUT
VIBRATION ACOUSTIC
ACCELERATION
VACUUM (5PSI)
VIBRATION TEMPERATURE
RECALIBRATION MAINTENANCE AND REPAIR
VIBRATION TEMPERATURE
SYSTEM NO. 5
FUNCTION CHECKOUT
VIBRATION (GROUND)
GROUND CHECKOUT
SHOCK (GROUND)
VIBRATION TEMPERATURE
VIBRATION ACOUSTIC
ACCELERATION
VACUUM (5PSI)
SYSTEM NO. 6
FUNCTION CHECKOUT
VIBRATION (GROUND)
GROUND CHECKOUT
VIBRATION TEMPERATURE
ACCELERATION
SHOCK (GROUND)
VACUUM (5PSI)
VIBRATION ACOUSTIC

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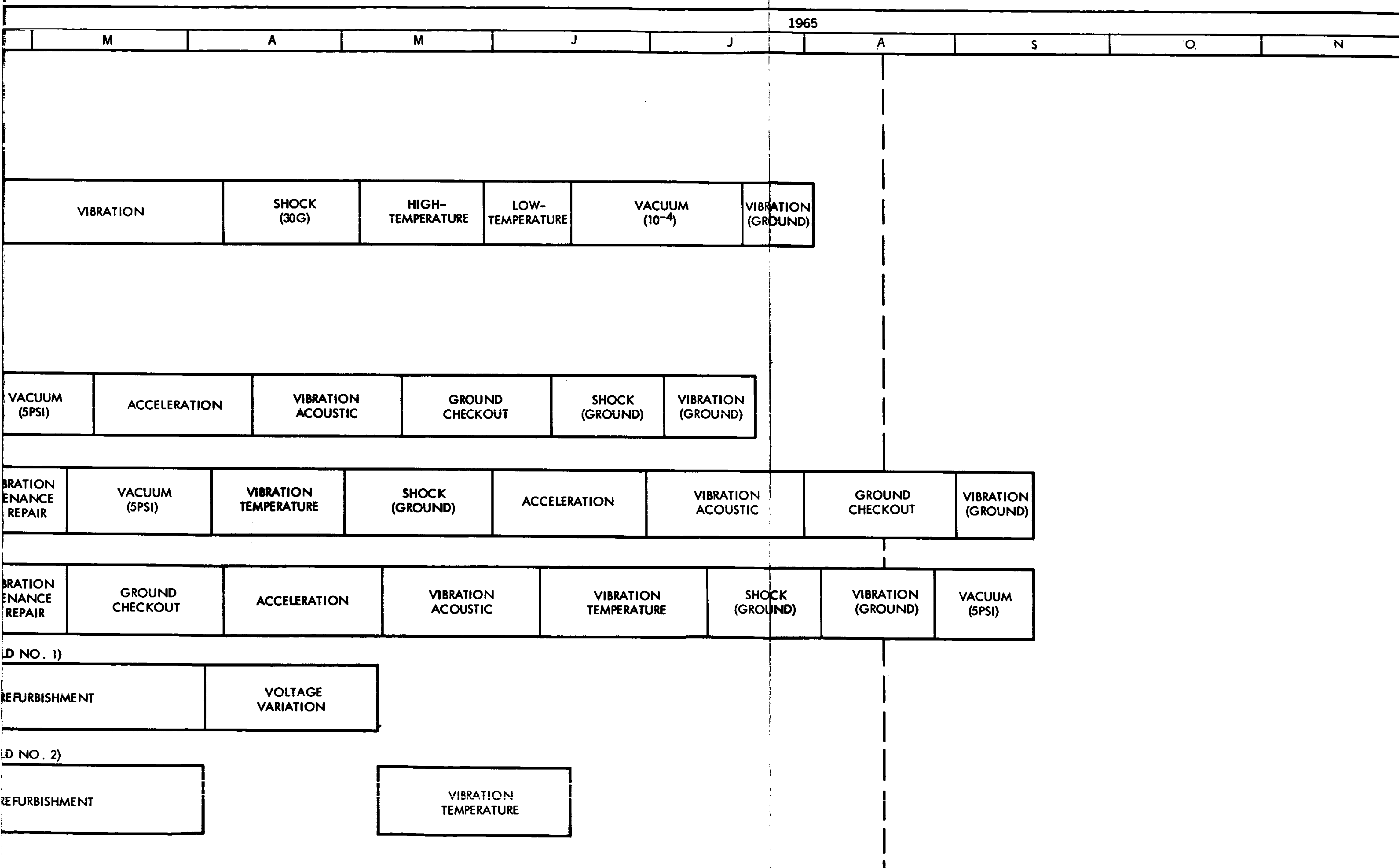

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Figure 5-33. SCS Qualification Test Schedule

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systems perform successfully after the subjection to the mission profile, first and third objectives, they are then considered qualified for use in flight qualification tests.

5.9.2 Reliability Assessment Model

5.9.2.1 Television Equipment and Central Timing Unit

The reliability assessment model for the television equipment will be submitted with the next quarterly revision of the Apollo General Test Plan. Figure 5-34 depicts the reliability assessment model for the CTU. The models will aid in establishing test continuity and in programming the collection of data for the assessment of reliability of this equipment by illustrating the reliability and functional interrelationships of the system components.

5.9.2.2 Communications and Data Subsystem

Figure 5-35 depicts the reliability assessment model for the telemetry of the communications and data subsystem. Figure 5-36 depicts the reliability assessment model for the C&D subsystem during the near-earth phase of the mission. Figure 5-37 depicts the reliability assessment model for the C&D subsystem during the deep-space phase of the mission. Figure 5-38 depicts the reliability assessment model for the C&D subsystem during recovery. Each subsystem is a series of the reliability function.

5.9.2.3 Antenna Systems

Figure 5-39 depicts the reliability assessment model for the recovery antenna system. Figure 5-40 depicts the reliability assessment model for the high-gain antenna system. The models will aid in establishing test continuity and in programming the collection of data for the assessment of reliability of the antenna systems by illustrating the reliability and functional interrelationships of each system's components.

5.9.3 Criticality and Hardware Requirements

The tables under this paragraph designate the criticality class, as defined in subsection 2.4, for the components of the subsystems listed. The table enumerates the hardware requirements for a minimum qualification program as established in subsection 2.5. The distribution of the hardware through the subsystem qualification test phases, as described in subsection 2.3, is shown. The location of the components in the spacecraft is indicated by spacecraft zone numbers as defined in subsection 4.2.

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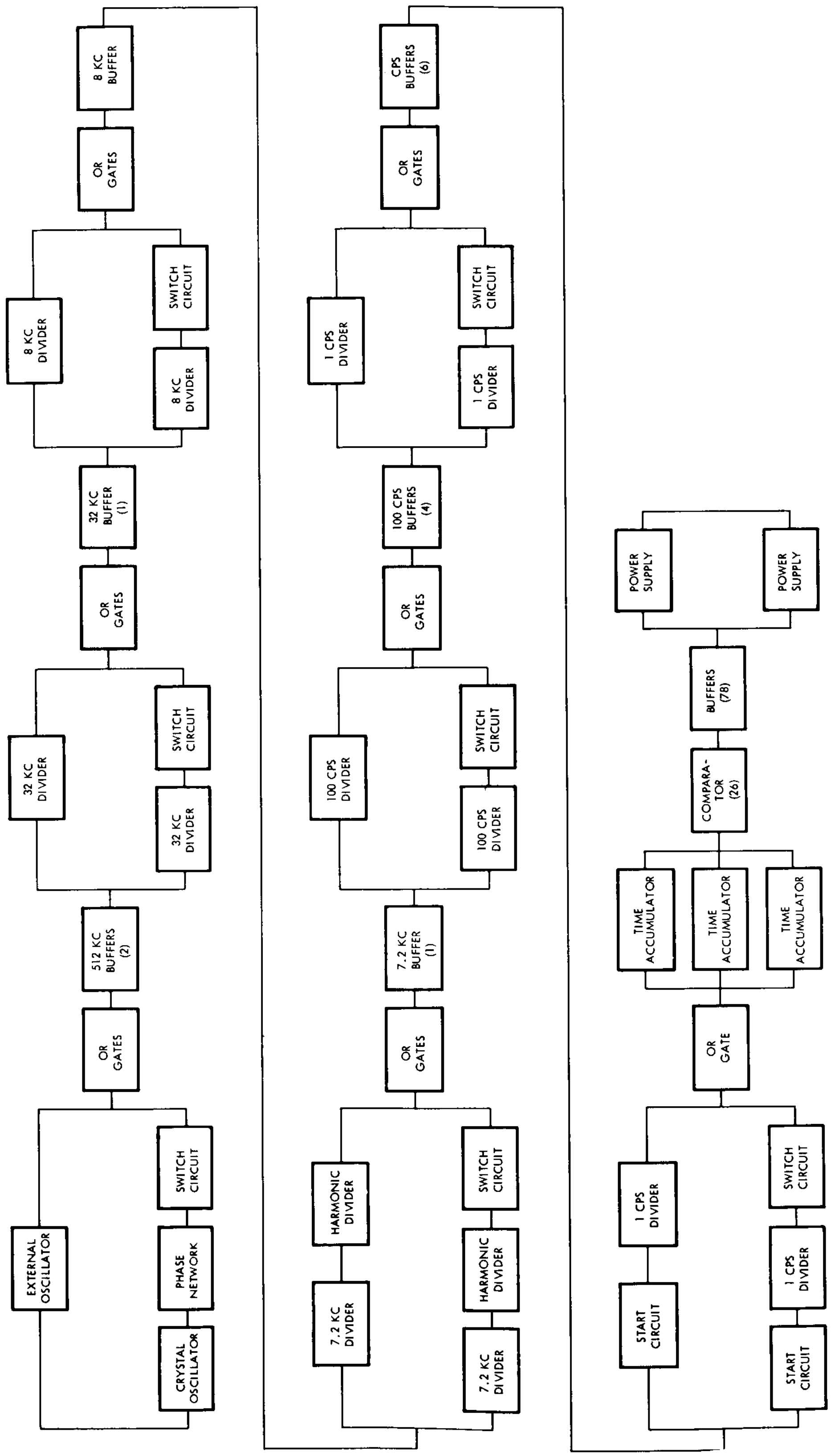

Figure 5-34. CTU Reliability Assessment Model

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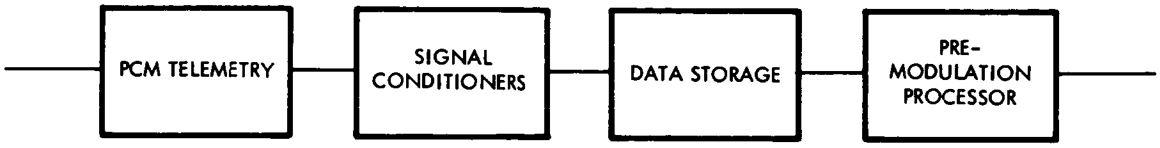

Figure 5-35. Telemetry Assessment Model

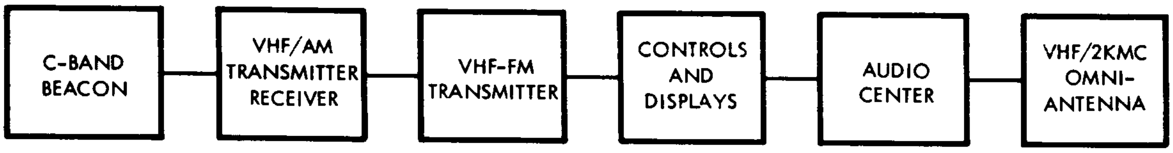

Figure 5-36. Near-Earth Assessment Model

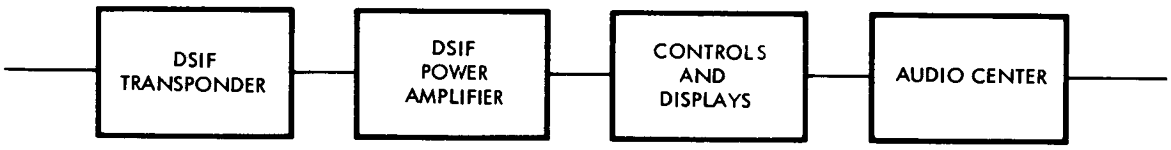

Figure 5-37. Deep-Space Assessment Model

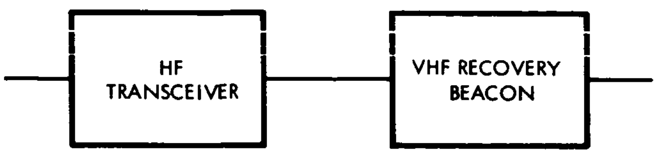

Figure 5-38. Recovery Assessment Model

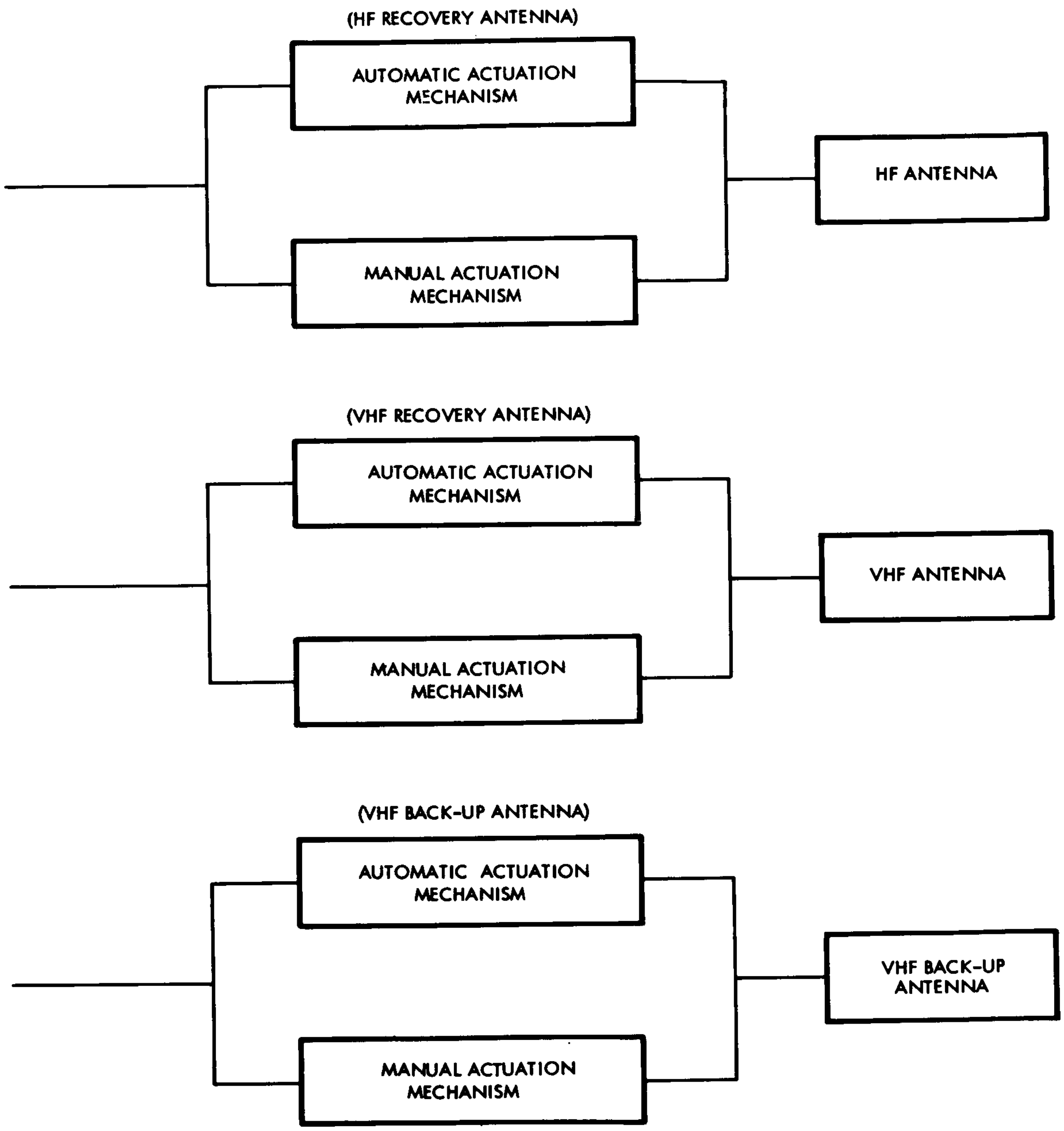

Figure 5-39. Recovery Antenna Assessment Model

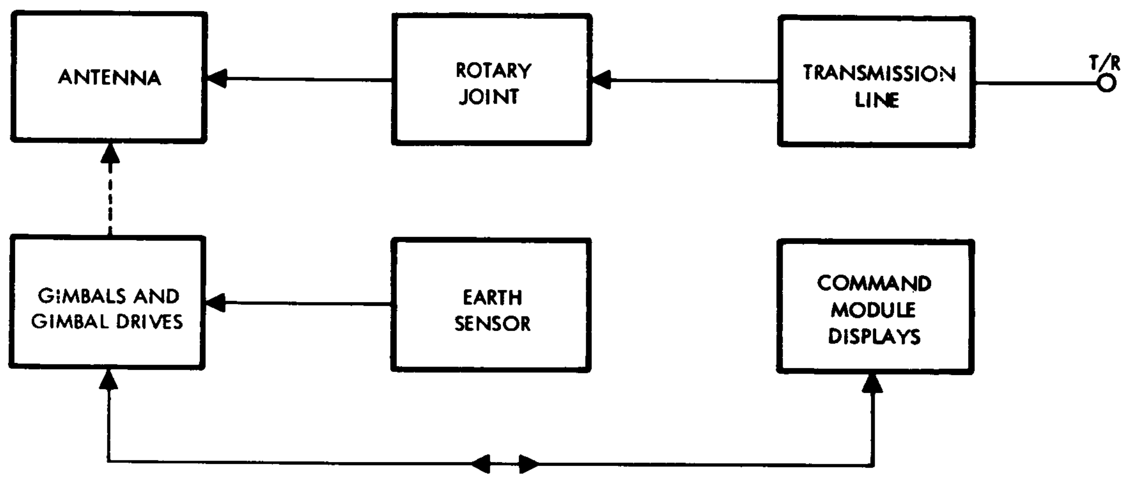

Figure 5-40. High-Gain Antenna Assessment Model

5.9.3.1 Television Equipment and Central Timing Unit

Table 5-18 exhibits the system parameters for qualification testing for the television equipment and CTU.

Table 5-18. System Parameters for Qualification Testing

Component	S/C Zone	Criticality	Required Hardware	Qualification Test Phase Distribution		
				A-1	A-2	B
TV equip.	2	3	4	2	2	1
CTU	2	2	6	2	2	3

5.9.3.2 Communications and Data Subsystem

Table 5-19 exhibits the system parameters for qualification testing of the communications and data subsystem.

Table 5-19. CDS Parameters for Qualification Testing

Component	S/C Zone	Criticality	Required Hardware	Qualification Test Phase Distribution		
				A-1	A-2	B
Audio center equipment	2	2	4	2	2	1
VHF/FM transmitter	2	3	6	3	2	2
VHF/AM transmitter-receiver	2	2	7	3	2	3
C-Band transponder	2	2	5	2	2	2
DSIF transponder	2	2	7	3	2	3
DSIF P. A.	2	2	7	3	2	3
VHF recovery beacon	2	3	5	3	2	1
HF transceiver	2	3	5	3	2	1
Pre-Mod. processor	2	2	7	3	2	3
Signal conditioners	2	3	5	2	2	2
Data storage	2	3	5	2	2	2
Comm. control panel	2	3	6	3	2	2
Audio control panel	2	3	6	3	2	2
PCM telemetry	2	3	5	3	2	1

5.9.3.3 Antenna Systems

Table 5-20 exhibits the system parameters for qualification testing of the antenna systems.

Table 5-20. Antenna Systems Parameters for Qualification Testing

Component	S/C Zone	Criticality	Required Hardware	Qualification Test Phase Distribution		
				A-1	A-2	B
HF recovery antenna	1	1	3	2	1	
VHF recovery antenna	1	3	3	2	1	
VHF backup recovery antenna	1	3	3	2	1	
High-gain antenna	3	2	6	2	1	3
C-Band antenna	1	2	3	2	1	

5.9.4 Qualification Test Schedule

The figures described in this paragraph display the qualification test schedules for the subsystems of the communications and instrumentation systems and define the test utilization of the hardware enumerated in the tables of subsection 5.9.3. The specific test levels to be applied in qualification testing are found in the tables of environmental test criteria in subsection 4.2 and are a function of the zone location of the equipment in the spacecraft. The zone numbers of the C&I subsystems and components are given in the tables associated with the test schedules described.

5.9.4.1 Television Equipment and Central Timing Unit

Figure 5-41 displays the qualification test schedule for the television equipment. Figure 5-42 displays the qualification test schedule for the central timing unit. Table 5-18 lists the associated system parameters.

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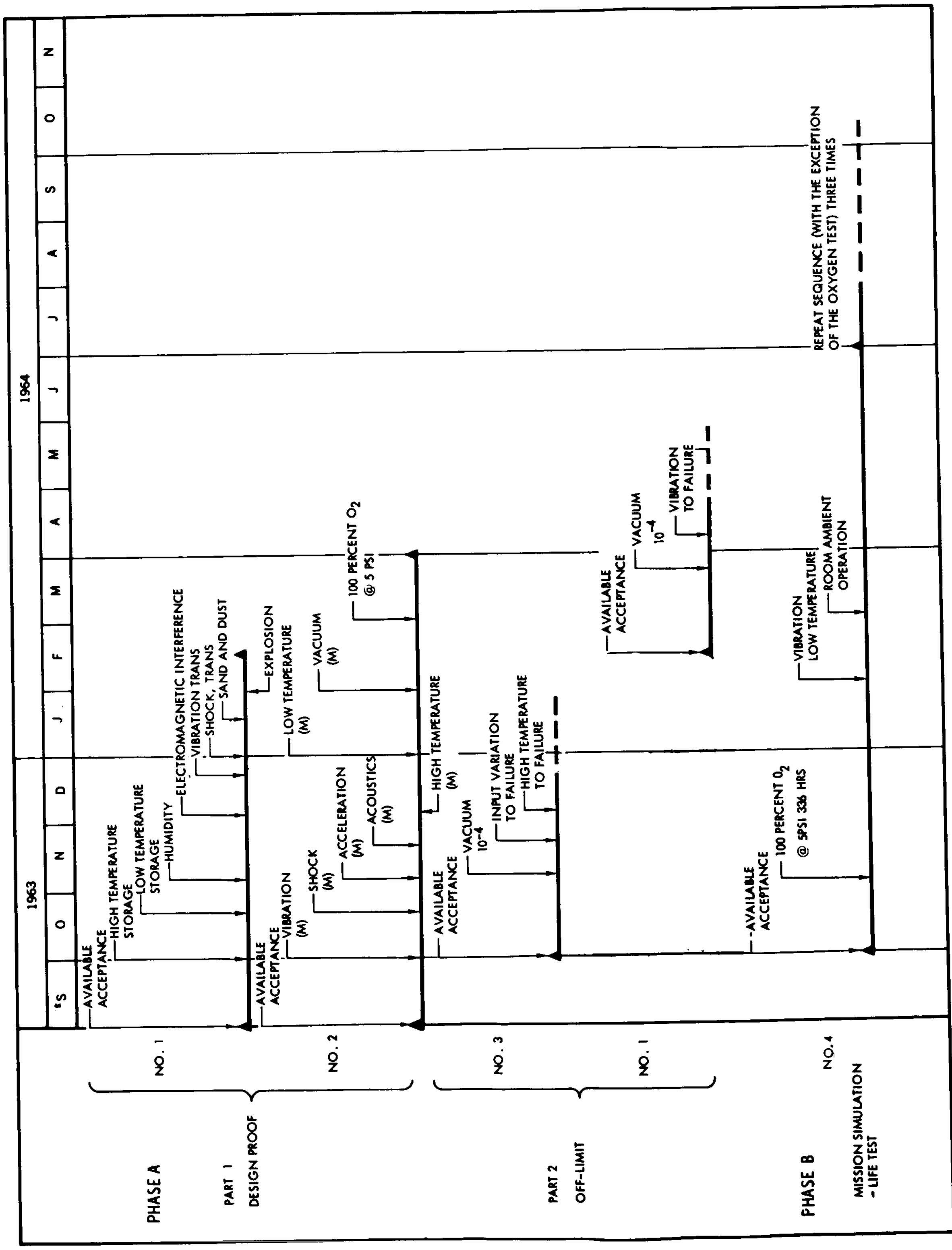

Figure 5-41 TV Equipment Qualification Test Schedule

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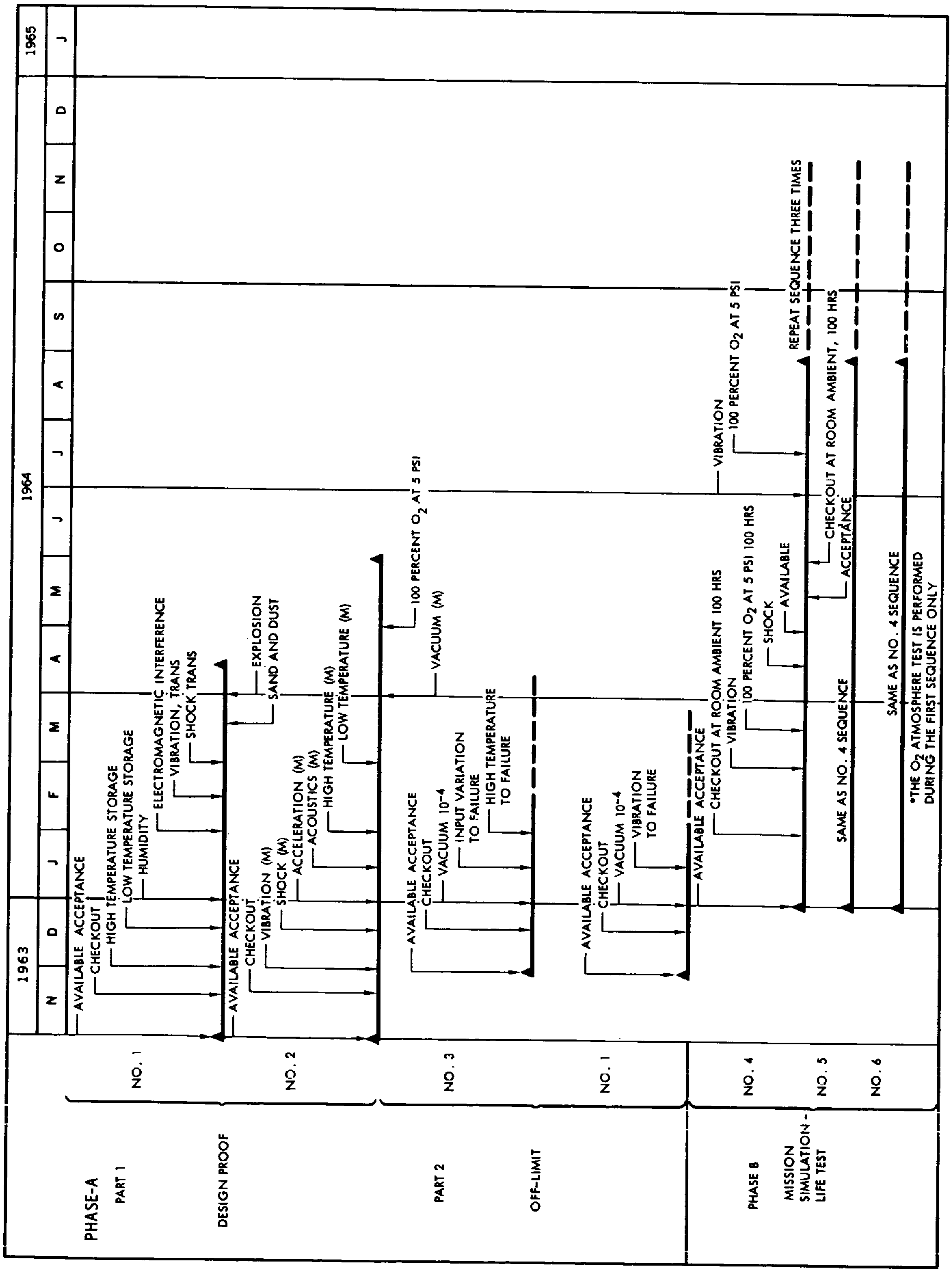

Figure 5-42. CTE Qualification Test Schedule

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5.9.4.2 Communications and Data Subsystem

Figure 5-43 displays the qualification test schedule for the CDS components of criticality category 2.

Figure 5-44 displays the qualification test schedule for the CDS components of criticality category 3.

Table 5-19 lists the associated system parameters for these test schedules.

5.9.4.3 Antenna Systems

Figure 5-45 displays the qualification test schedule for the HF recovery antenna.

Figure 5-46 displays the qualification test schedule for the VHF recovery antenna.

Figure 5-47 displays the qualification test schedule for the VHF recovery (back-up) antennas.

Figure 5-48 displays the qualification test schedule for the 2KMC high-gain antenna.

Figure 5-49 displays the qualification test schedule for the C-band beacon antenna and incorporates the antenna window qualification.

Table 5-20 lists the associated system parameters for these test schedules.

5.10 LIFE SYSTEMS

5.10.1 Scope

Since many components will be qualified and furnished by NASA, life system tests will be conducted to verify compatibility of crew equipment and related GSE systems machine interfaces. Qualification testing will be conducted with AFRM vehicle mock-ups and production equipment.

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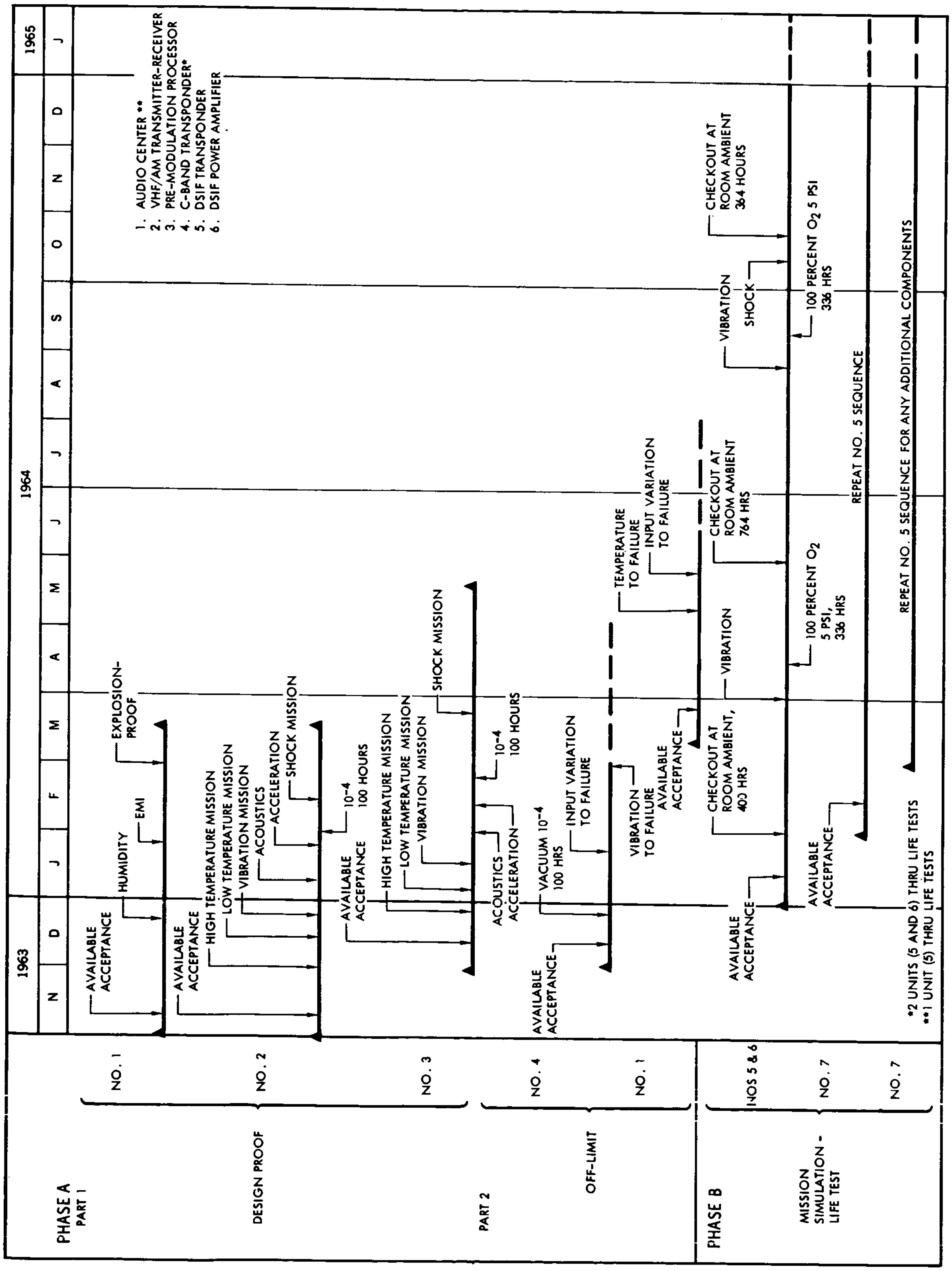

Figure 5-43. CDS Qualification Test Schedule (Criticality 2)

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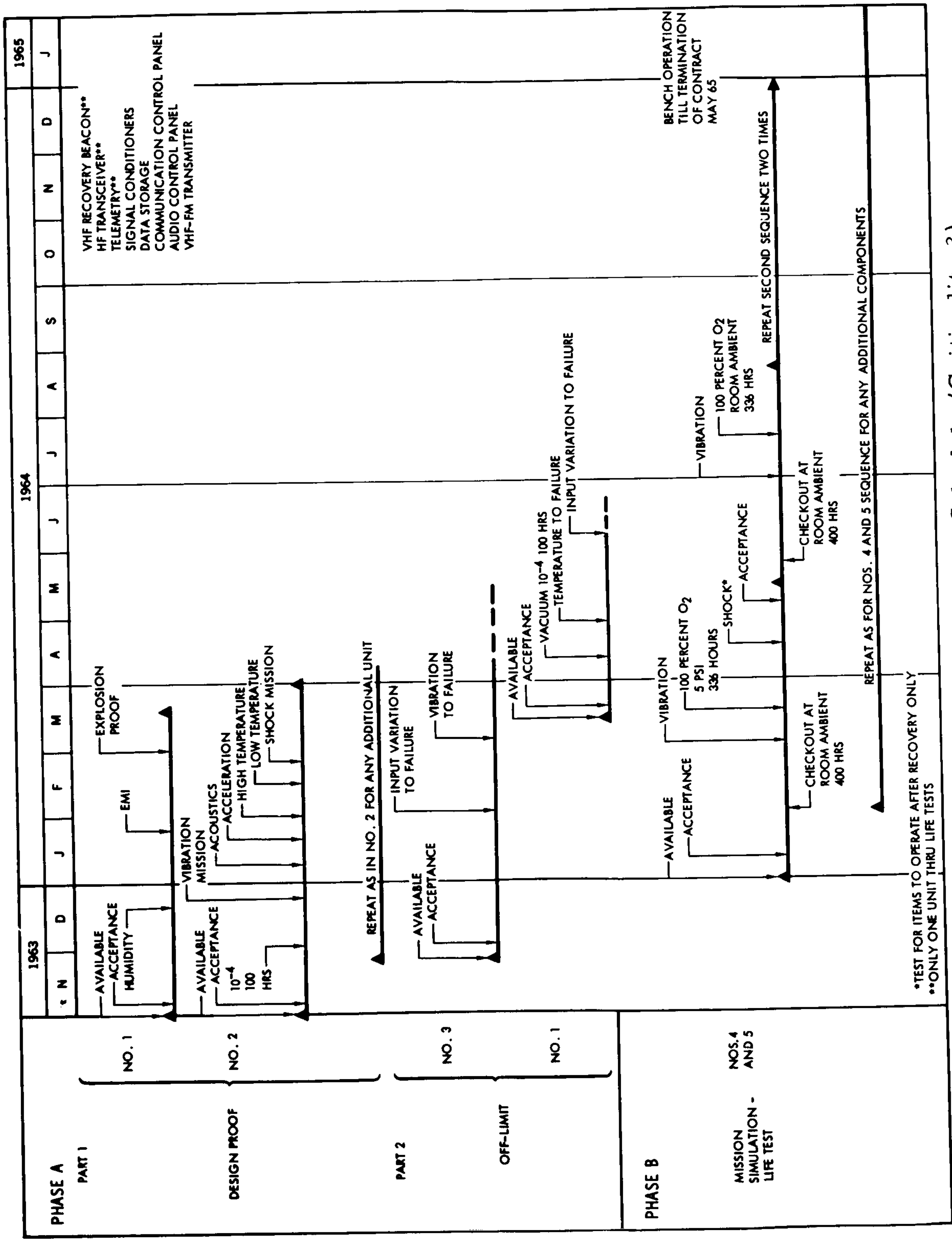

Figure 5-44. CDS Qualification Test Schedule (Criticality 3)

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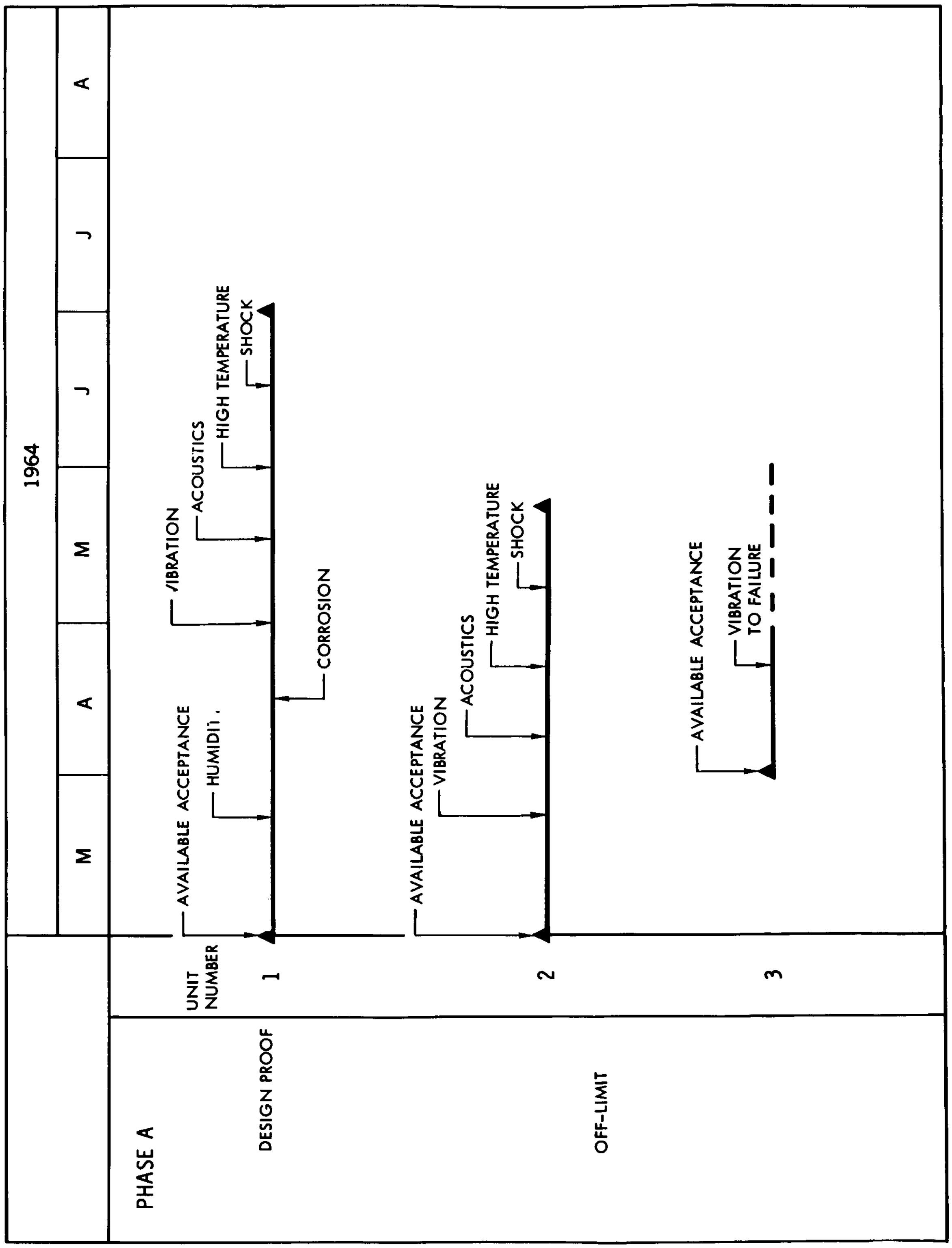

Figure 5-46. VHF Recovery Qualification Test Schedule

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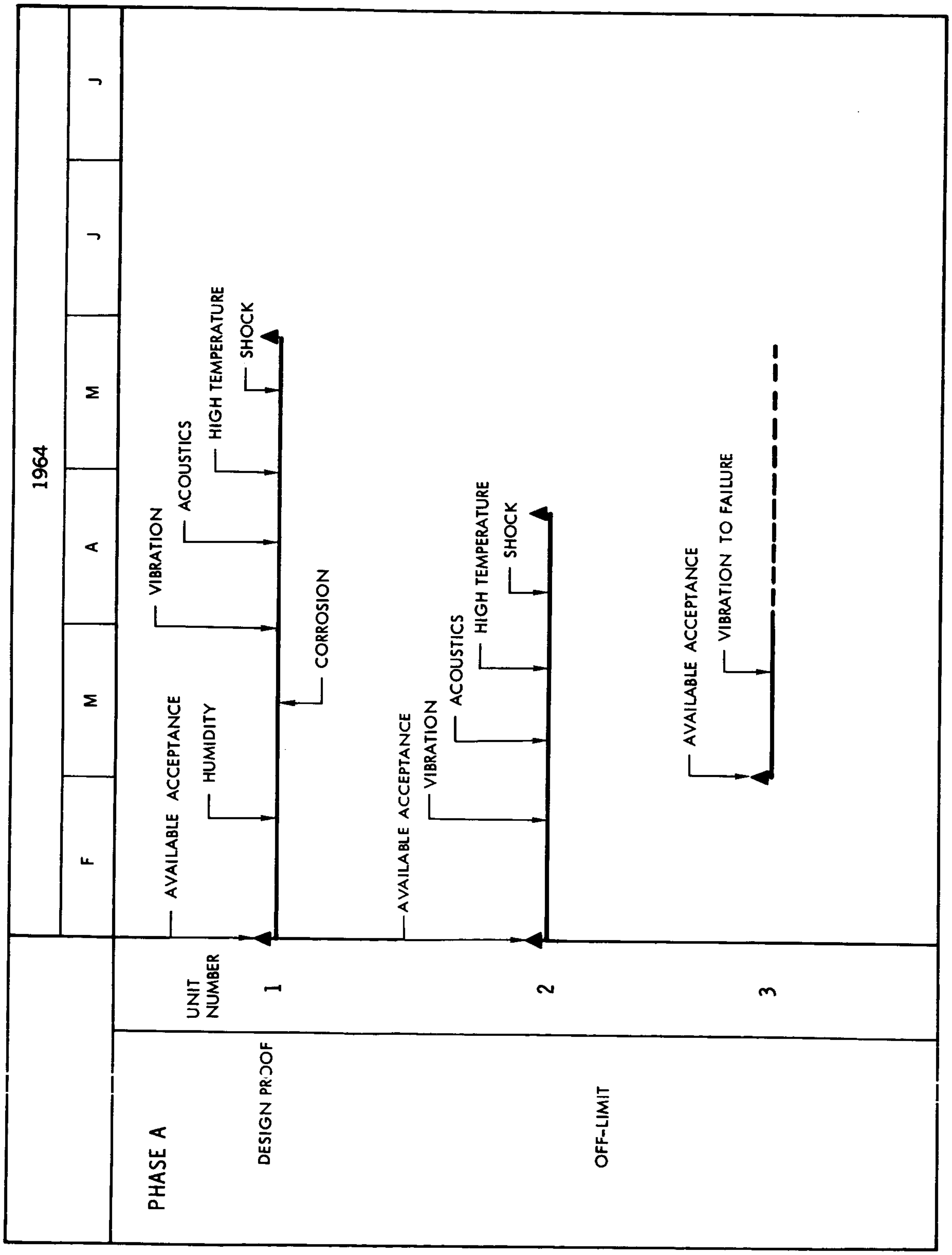

Figure 5-47. VHF Recovery (Back-up) Qualification Test Schedule

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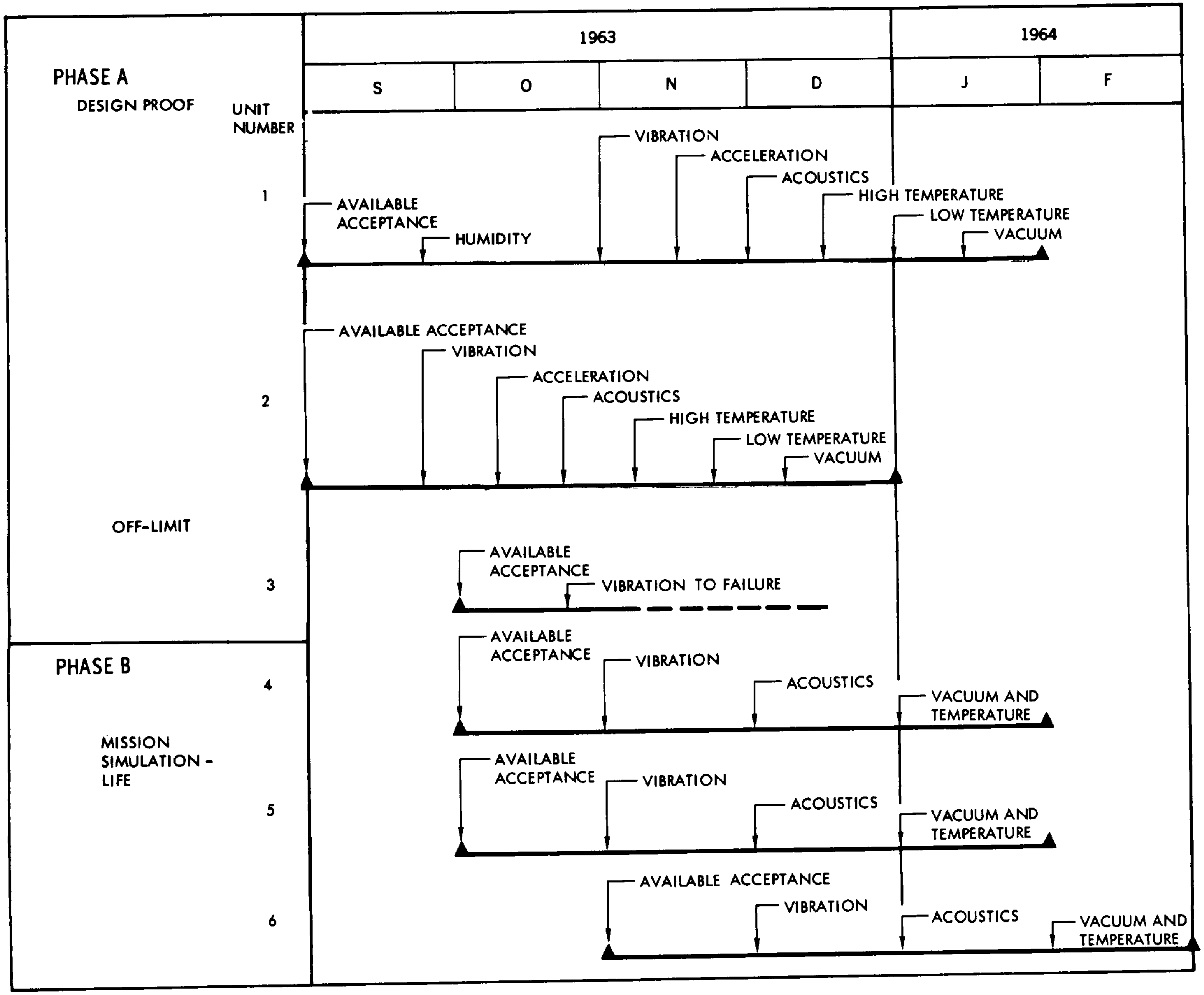

Figure 5-48. Two-kmc High-Gain Qualification Test Schedule

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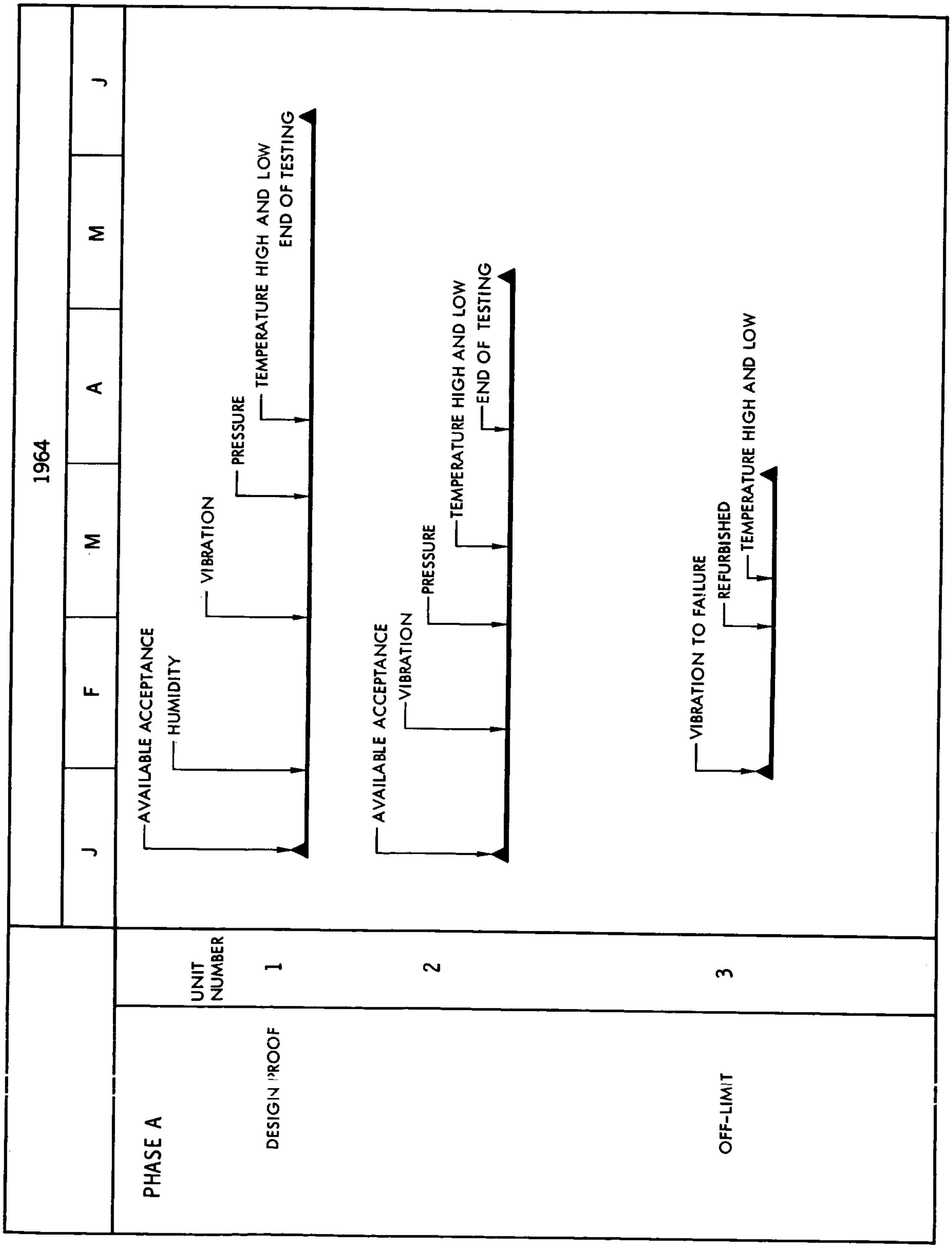

Figure 5-49. C-Band Beacon Qualification Test Schedule

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5.10.2 Reliability Assessment Model

Due to the number of components and the varied nature of the qualification testing, a reliability assessment model is not considered applicable for the life systems.

5.10.3 Criticality and Hardware Requirements

Table 5-21 designates the criticality class, as defined in Section 2.4, for the components of the life systems. The table enumerates the hardware requirements for a minimum qualification program, as established in Section 2.5. The location of the components in the spacecraft is indicated by spacecraft zone numbers, as defined in Section 4.2.

5.10.4 Qualification Test Schedule

Figure 5-50 displays the qualification test schedule for the components of the life systems. Because the particular tests to be applied are not defined at this time, the schedule displays the time phasing of the test categories, only.

5.11 THERMAL PROTECTION SYSTEM

5.11.1 Scope

The ground qualification test plan for the thermal protection system (TPS) consists of environmental qualification and mission simulation-life test of heat shield panels fabricated with the same ablative and bonding materials and in the same configuration as that which will be used for the C/M heat shield. The data obtained will be used for backup to the C/M heat shield flight qualification testing and for assessment of TPS reliability.

5.11.2 Reliability Assessment Model

Figure 5-51 illustrates the functional relationship and the series reliability configuration of the heat shield materials.

5.11.3 Criticality and Hardware Requirements

Table 5-22 designates the criticality class, as defined in Section 2.4; the number of panels required for an optimized ground qualification test plan, as described in Section 2.5; and the zone locations of the heat shield in the spacecraft, as defined in Section 4.2. The distribution of the panels through the qualifications test phases, as described in Section 2.3, is shown.

[redacted]

Table 5-21. Life Systems Parameters for Qualification Testing

Component	S/C Zone	Criticality*	Required Hardware
Tool set in-flight	2		4
Pad assembly couch	2		6
Harness assembly couch	2		4
Restraint assembly rest station	2		4
Portable light assembly	2		8
Constant wear garment	2		4
Hose umbilical	2		8
Disconnect assembly cabin half	2		6
Food and food package	2		8
Canister assembly – fecal	2		4
Bag (set) fecal	2		4
Receptacle assembly crewmen relief	2		4
Receptacle, liner set	2		3
Head set and microphone	2		6
Transceiver	2		6
Cleansing pad	2		3
Ingestible dentifrice	2		3
Shaver	2		3
Deodorizer pad	2		3
Emergency medical supplies and equipment	2		3

*To be determined

[redacted]

[redacted]

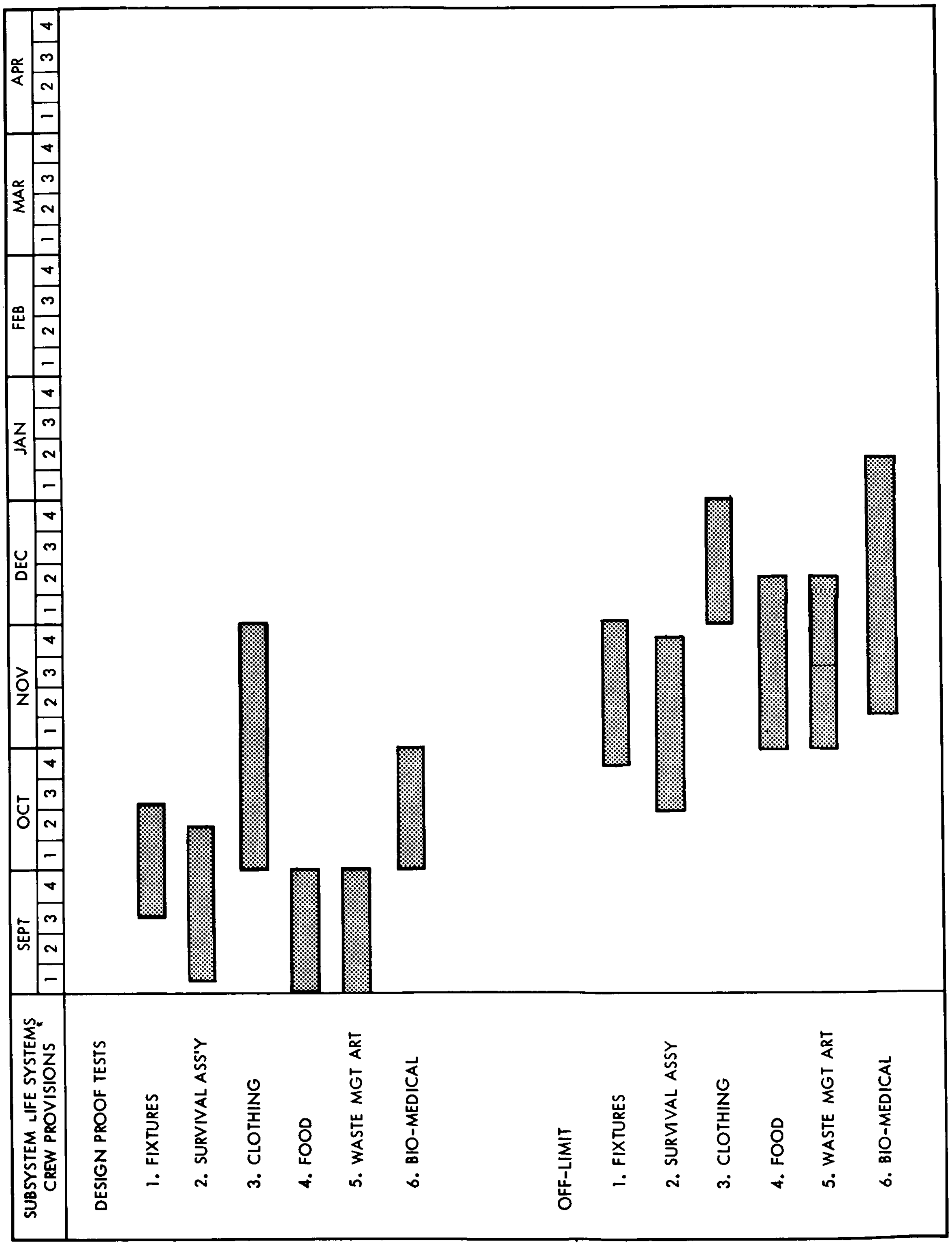

Figure 5-50. Life Systems Qualification Test Schedule

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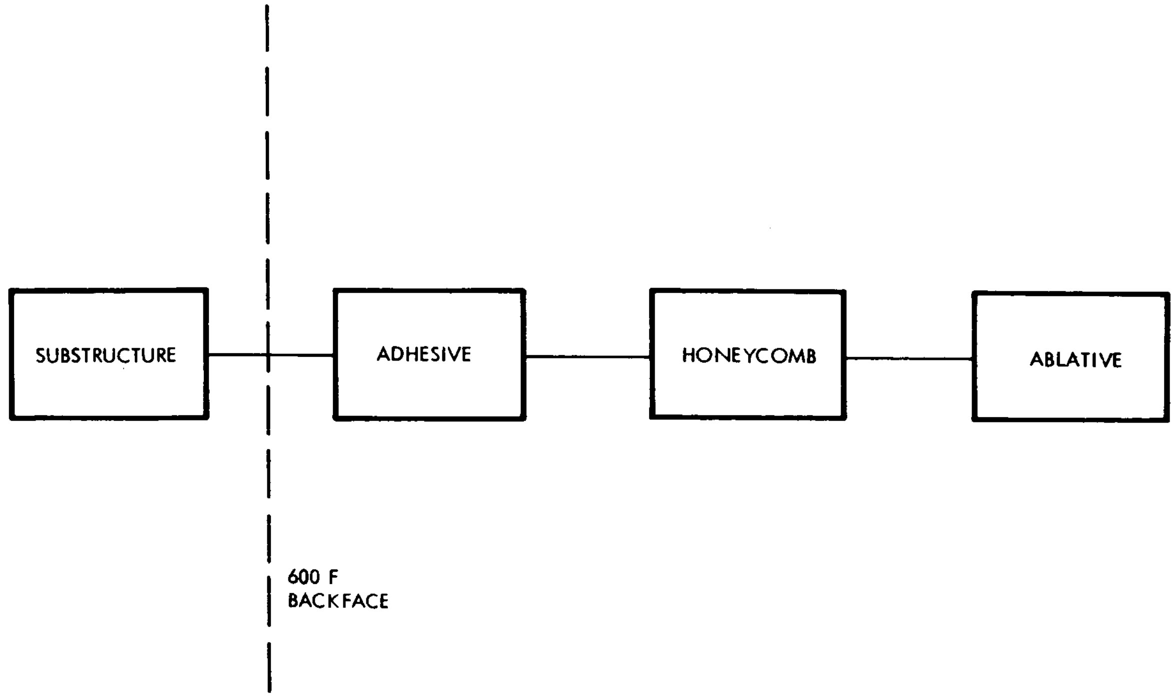

Figure 5-51. TPS Reliability Assessment Model

Table 5-22. TPS Parameters for Qualification Testing

Component	S/C Zone	Criticality	Required Hardware	Qualification Test Phase Distribution		
				A-1	A-2	B
Heat shield panel	1	1	16	6	4	6

5.11.4 Qualification Test Schedule

Figure 5-52 displays the qualification test schedule for the TPS and defines the test utilization of the panels enumerated in Table 5-22. The specific test levels to be applied in qualification testing are found in the tables of environmental test criteria in Section 4.2 and are a function of the zone location of the heat shield in the spacecraft. The zone number is given in Table 5-22.

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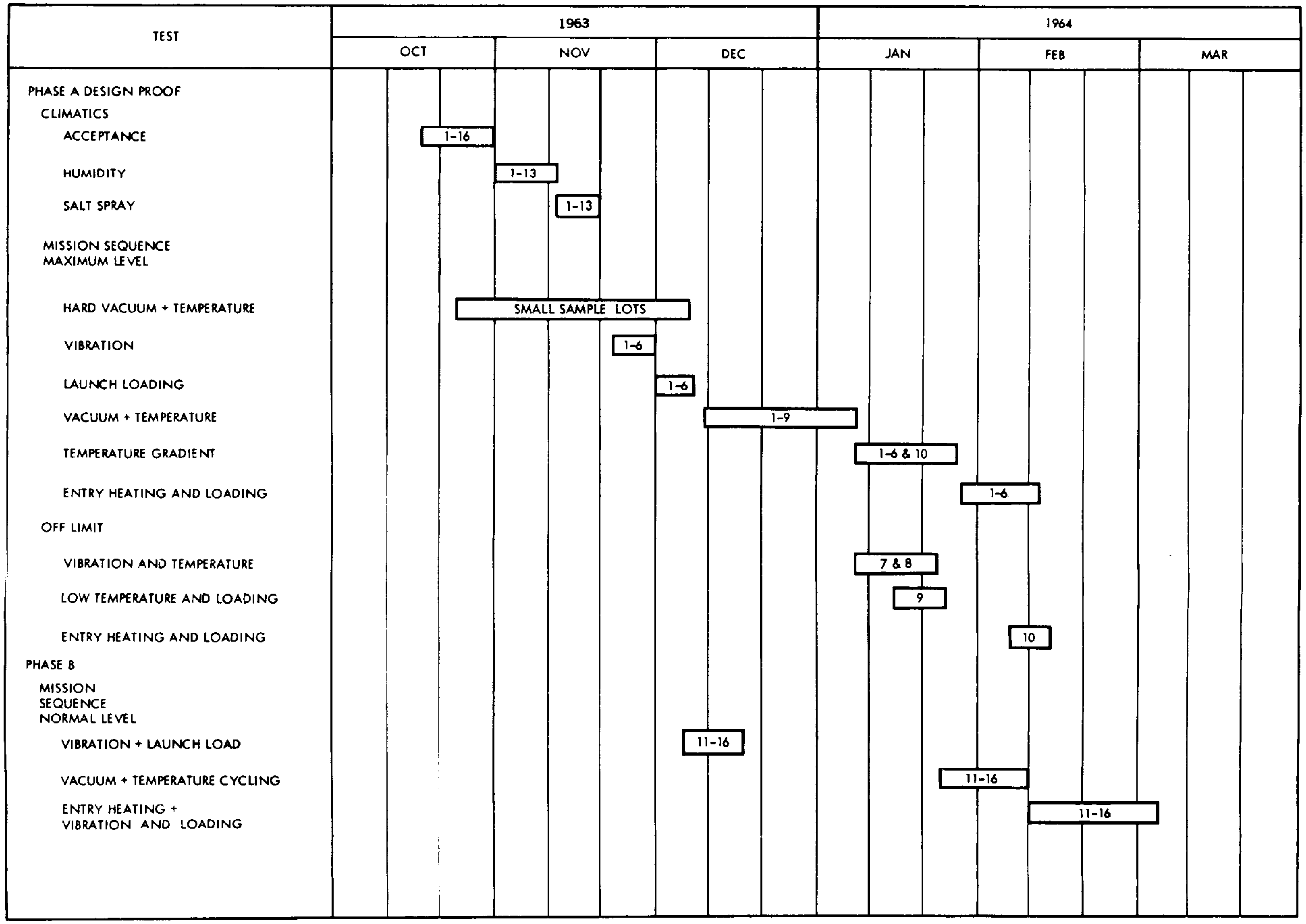

Figure 5-52. TPS Qualification Test Schedule

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5.12 STRUCTURAL SYSTEM

5.12.1 Scope

The qualification test program includes testing of structural components and complete structures of command module, service module, spacecraft adapter, launch escape tower, and combined structures. Subassembly testing of structural members including joints, truss sections, welds, tankage, windows, and motor supports, will be conducted by applying limit loads for major critical design conditions and then by applying ultimate design loads. The test specimen will be loaded to failure for a selected condition for each test to determine the maximum load capability.

5.12.2 Reliability Assessment Model

Problems in the reliability analysis of structural components and complete structural assemblies including the development of reliability assessment models are being studied. When presentation techniques have been developed, a reliability assessment model will be included in the structures qualification test plan.

5.12.3 Criticality and Hardware Requirements

The criticality and hardware requirements of the structural system are being determined and will be included in a later revision of this volume.

5.12.4 Qualification Test Schedule

The structures system qualification test schedule will be defined to the extent of information available and will be submitted in later revisions.

5.13 MECHANICAL SYSTEMS

5.13.1 Scope

The qualification test program includes component and systems testing of the four separation systems and the crew hatch and astro sextant door actuating systems, and component testing of the crew couch attenuation struts.

5.13.1.1 Forward Heat Shield Separation System

The separation of the forward heat shield from the command module, prior to operation of the earth landing system, will be accomplished by firing two pressure cartridges. The gas pressure operates four thrusters that will each break one tie bolt. The qualification test program for the

[redacted]

[redacted]

separation system will be included in a later revision. The qualification test program for the pressure cartridge will subject 118 specimens to mission environments and firing tests.

5.13.1.2 Launch Escape Tower Separation System

The separation of the launch escape tower from the command module will be accomplished by firing four explosive bolts. Each bolt has two independent cutting modes, either of which will separate the bolt. The qualification test program for the separation system will be included in a later revision. The qualification test program for the bolts will subject 133 specimens to mission environments, loads, and firing tests.

5.13.1.3 Command Module to Service Module Separation System

The command module will be separated from the service module by firing detonators and linear-shaped charges in three tie rod assemblies. Each tie rod assembly has two detonators and two linear-shaped charges in redundant pairs. The qualification test program for the separation system will be included in a later revision. The qualification test program for the detonator will subject 118 specimens to mission environments and firing tests.

5.13.1.4 Adapter Separation System

The service module will be separated from the launch vehicle upper stage by firing detonators and linear-shaped charges. The cylindrical adapter for the S-I vehicle and the tapered adapter for the S-V vehicle will have basically similar systems. The explosive separation system consists of 40 lengths of flexible linear-shaped charge, 8 interface junction blocks, and 2 umbilical cable cutters, each with 2 lengths of linear-shaped charge and 4 lengths of mild detonating fuse. The system is initiated by two detonators, one of which is redundant. The qualification test program for the separation system will be included in a later revision. The qualification test program for the linear-shaped charge will subject 100 specimens to mission environments and penetration tests; the detonator test program is referred to in 5.13.1.3.

5.13.1.5 Crew Hatch Actuating System

The crew hatch will be operated manually from inside the crew compartment. A hermetically sealed, manually operated actuator will operate latches on the outside of the inner crew hatch of the command module. The qualification test program for the complete actuating system will be included in a later revision. The qualification test program for the manually operated actuator will subject four specimens to mission environments, loading, and life tests.

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5.13.1.6 Astro Sextant Door Actuating System

The doors covering the astro sextant window will be operated manually from inside the crew compartment. A hermetically sealed, manually driven gear box will apply torque to three rotary flexible shafts. Two of the shafts will operate the door mechanism and the third will actuate the latch mechanical linkage by means of a screwjack. The qualification test program for the complete actuating system will be included in a later revision. The qualification test programs for the gear box, rotary flexible shaft, and screwjack will subject four specimens of each to mission environments, loading, and life tests.

5.13.1.7 Crew Couch Shock Attenuation System

The shock loads induced in the crew couches by earth landing will be attenuated by a total of eight shock struts of four different types. The qualification test program for the complete attenuation system will be included in a later revision. The qualification test program for the shock struts will subject seven specimens of each type of strut to mission environments, and normal and emergency impact loading. Each strut will be subjected to twenty impacts to compress the replaceable crushable honeycomb core.

5.13.2 Reliability Assessment Model

5.13.2.1 Forward Heat Shield Separation System

Figure 5-53 depicts the reliability assessment model for the forward heat shield separation system pressure cartridges.

5.13.2.2 Launch Escape Tower Separation System

Figure 5-54 depicts the reliability assessment model for the LET separation system explosive bolts.

5.13.2.3 Command Module to Service Module Separation System

Figure 5-55 depicts the reliability assessment model for the C/M to S/M separation system.

5.13.3. Criticality and Hardware Requirements

Table 5-23 designates the criticality class, as defined in Section 2.4, for the components of the mechanical devices. The table enumerates the hardware requirements for a minimum qualification program, as established in Section 2.5. The distribution of the hardware through the subsystem

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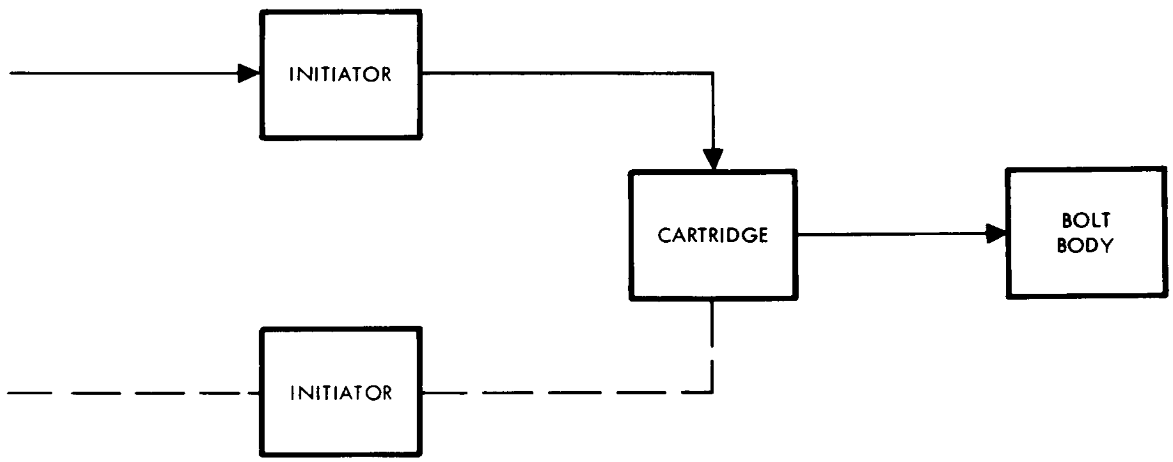

Figure 5-53. Pressure Cartridge Reliability Assessment Model

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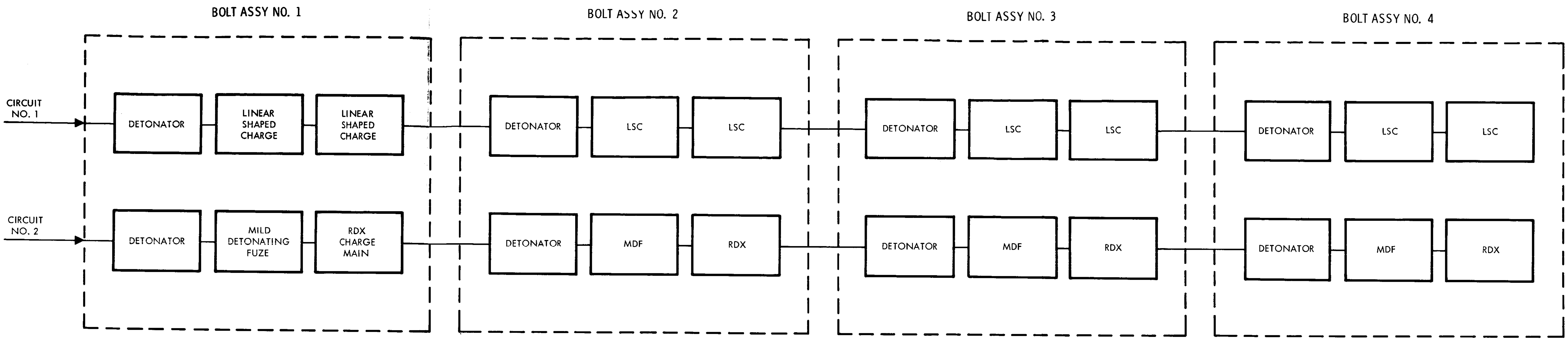
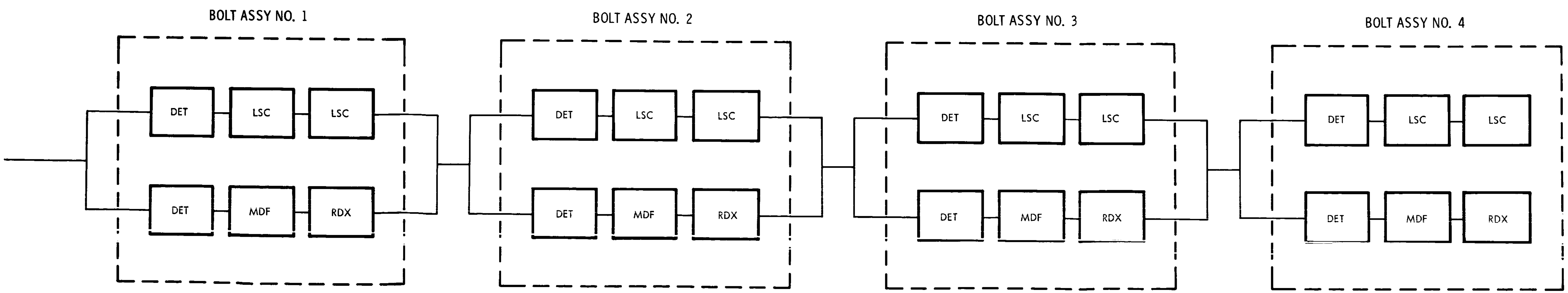


TEST MODES: NO. 1 CIRCUIT ONLY
NO. 2 CIRCUIT ONLY



TEST MODE: NO. 1 & NO. 2 CIRCUITS TOGETHER

Figure 5-54. Reliability Assessment Models Launch Escape Tower Separation System

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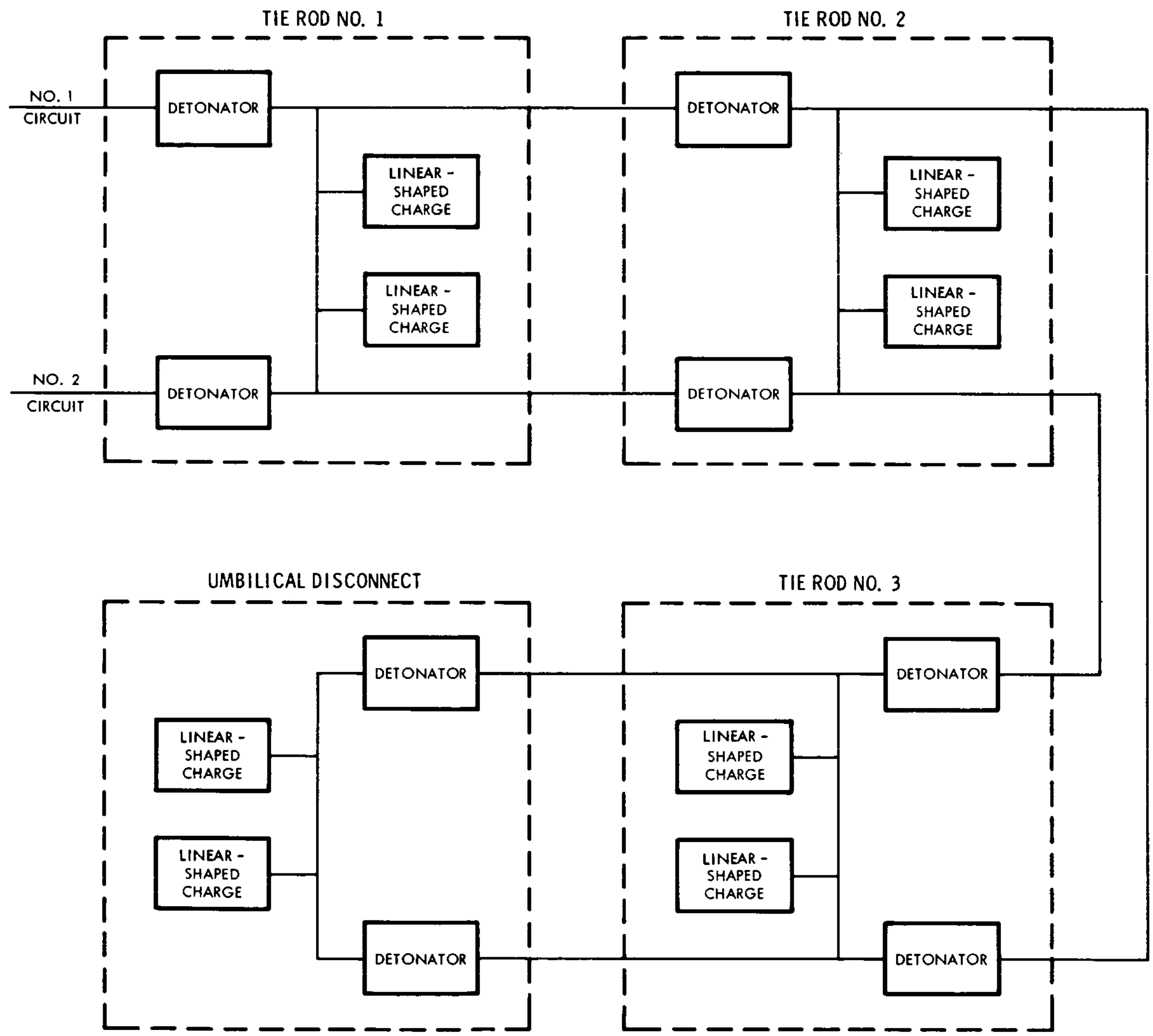

TEST MODES: NO. 1 CIRCUIT ONLY
NO. 2 CIRCUIT ONLY
NO. 1 & NO. 2 CIRCUITS TOGETHER

Figure 5-55. Reliability Assessment Model C/M to S/M Separation System

 [redacted]

[redacted] [redacted]

qualification test phases, as described in Section 2.3, is shown. The location of the components in the spacecraft is indicated by spacecraft zone numbers, as defined in Section 4.2.

Table 5-23. System Parameters for Qualification Testing

Component	S/C Zone	Criticality	Required Hardware
Cartridge, forward compartment heat shield separation	1	1	118
Explosive bolt, launch escape tower separation	1	1	133
Detonator cartridge	1	1	118
Crew hatch actuator	1 & 2	3	4
Astro sextant door gear box	1 & 2	1	4
Astro sextant door rotary flexible shaft	1 & 2	1	4
Astro sextant door screwjack	1	1	4
Crew couch shock struts	2	1	7 of four types
Linear shaped charge	1, 3, 4	1	100 of each type
Crew hatch actuating system	1 & 2	3	2
Astro sextant door actuating system	1 & 2	1	2

5.13.4 Qualification Test Schedule

5.13.4.1 Forward Heat Shield Separation System

Figure 5-56 displays the qualification test schedule for the cartridges enumerated in Table 5-23. The specific test levels to be applied in qualification testing are found in the tables of environmental test criteria in Section 4.2 and are a function of the zone location of the equipment in the spacecraft. The zone number is given in Table 5-23.

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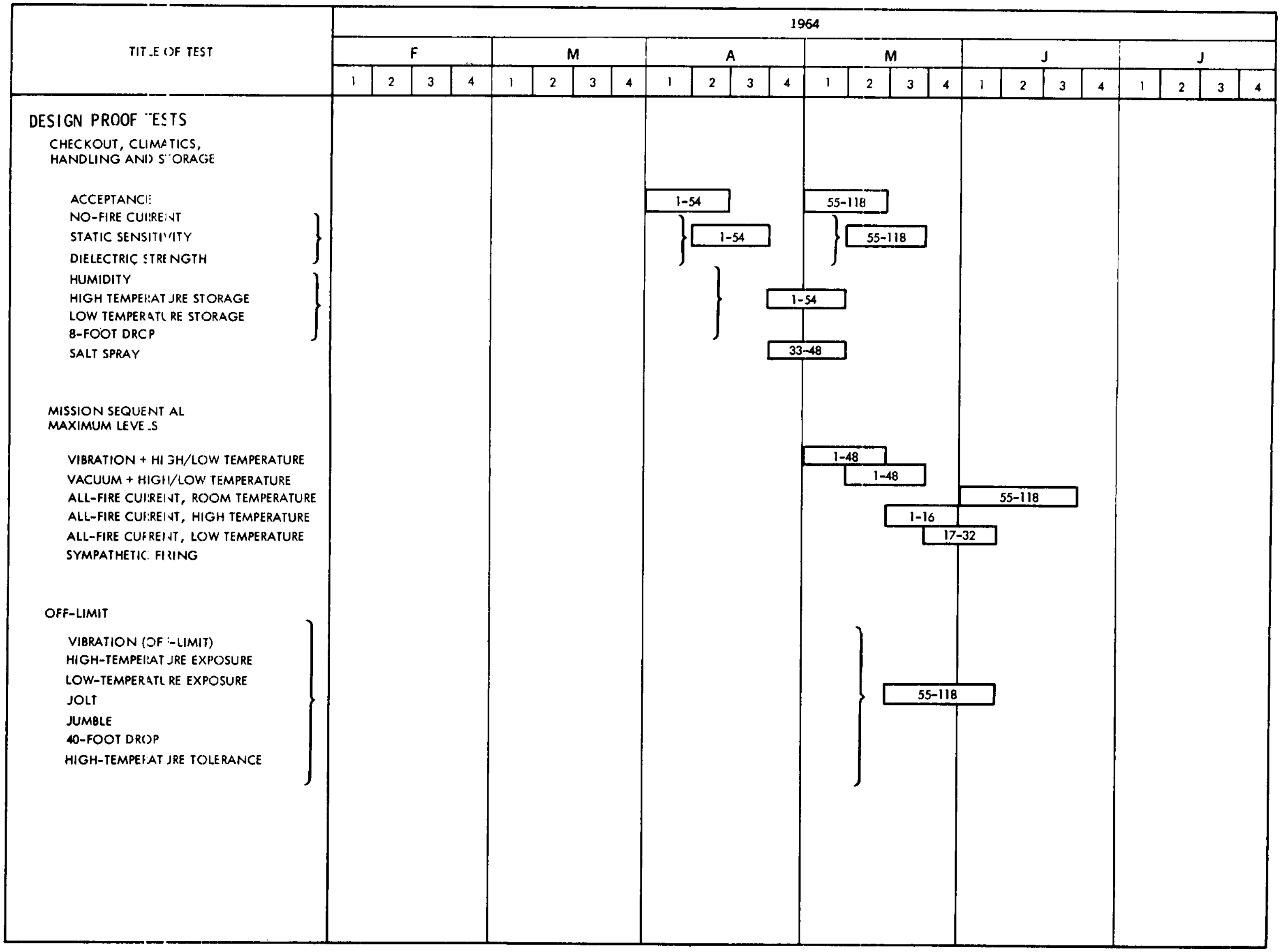

Figure 5-56. Pressure Cartridge Qualification Test Schedule

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5.13.4.2 Launch Escape Tower Separation System

Figure 5-57 displays the qualification test schedule for the LET explosive bolts enumerated in Table 5-23. The specific test levels to be applied in qualification testing are found in the tables of environmental test criteria in Section 4.2 and are a function of the zone location of the equipment in the spacecraft. The zone number is given in Table 5-23.

5.14 ELECTRONIC SYSTEMS

5.14.1 Scope

The electronic systems (ES) consist of the display and controls system (D&C) and the in-flight test system (IFTS). The D&C system will be tested at three levels of assembly: part, subpanel assembly, and the control console.

The parts qualification program will consist of a complete environmental and stress testing program. Those parts which are standardized in several subpanel usage applications will be defined as criticality one parts, in order to achieve maximum test data from a smaller total sample size of each standardized part. This philosophy applies to parts such as meters, switches, wire, and connectors used in D&C subpanels of varying criticalities. Nonstandard parts will be qualified according to their particular criticality requirements.

At the subpanel level, the qualification program will include acceptance type tests, operational and wiring checkouts, some off-limit tests, and a modified Phase B or Mission Simulation-Life level test program.

A complete control console will be assembled and tested through the dynamic and operational test phases. A combined vacuum-temperature operating test and temperature-vibration test also will be run on the control console. At the conclusion of all other testing, an off-limit shock test will be performed.

The supplier will qualify parts; S&ID will qualify the subpanels assembled in house and the control console.

The IFTS will be qualified by the subcontractor.

5.14.2 Reliability Assessment Model

The multiplicity and separate purpose of the subpanel assemblies of the control console do not lend themselves to the drawing of a comprehensive reliability assessment model. Therefore, Figure 5-58 depicts the reliability assessment model for the IFTS only. The individual system subpanels are

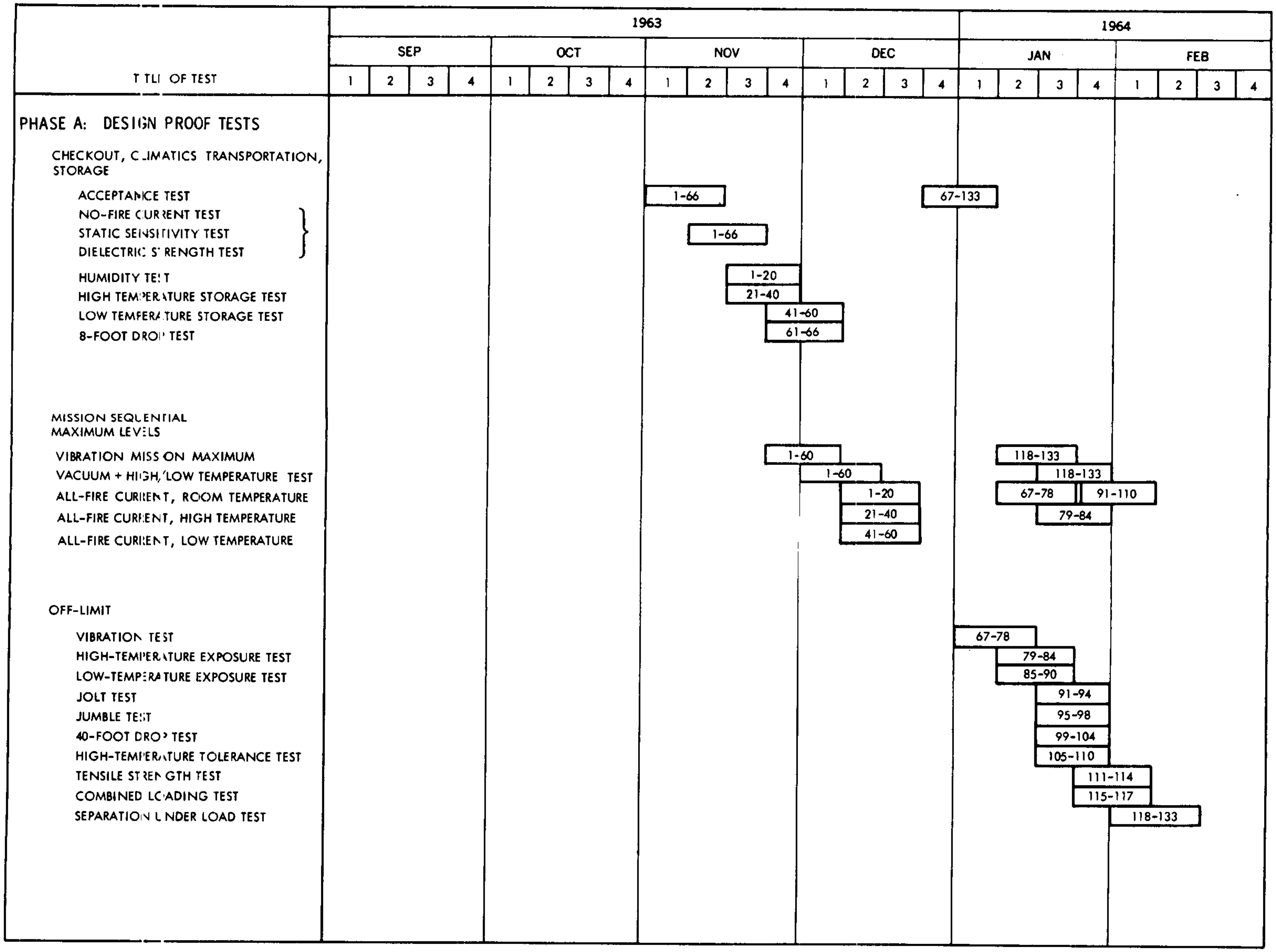

Figure 5-57. Explosive Bolt Qualification Test Schedule

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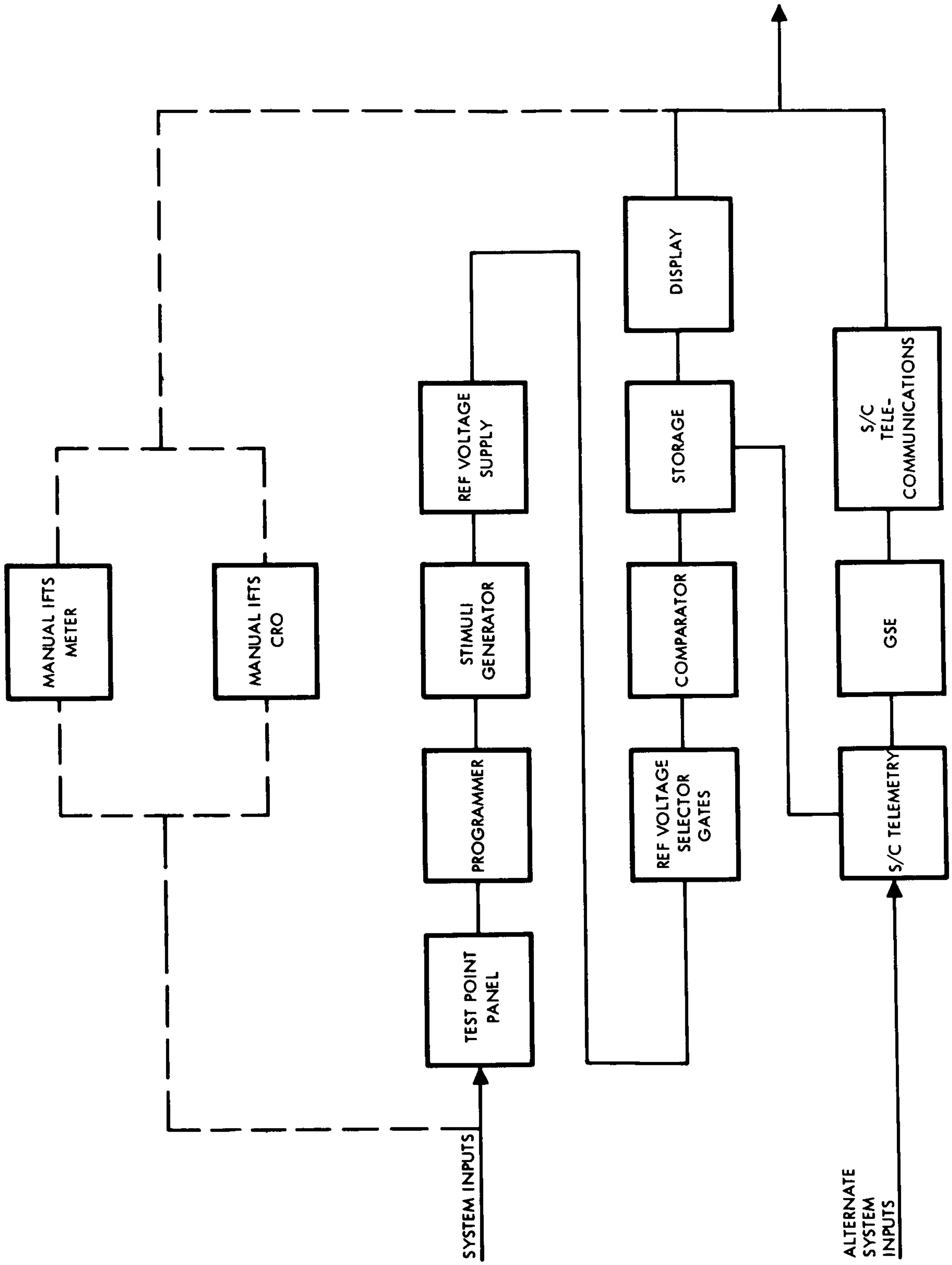

Figure 5-58. IFTS Reliability Assessment Model

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a part of that system model and are used as such. The IFTS model will aid in establishing test continuity and in programming the collection of data for the assessment of reliability by illustrating the functional and reliability interrelationships of the system components.

5.14.3 Criticality and Hardware Requirements

Table 5-24 designates the criticality class, as defined in Section 2.4, for the components of the ES. The table enumerates the hardware requirements for a minimum qualification program, as established in Section 2.5. The distribution of the hardware through the subsystem qualification test phases, as described in Section 2.3, is shown. The location of the components in the spacecraft is indicated by spacecraft zone numbers, as defined in Section 4.2. This table will be completed as information becomes available.

5.14.4 Qualification Test Schedule

Figure 5-59 displays the qualification test schedule for the D&C subpanel assemblies and control console, and defines the test utilization of the hardware enumerated in Table 5-24. Subpanel assemblies numbered 2, 4, 6, 7, 9, and 14 will be qualified at the subpanel level by the vendor or subcontractor at his own facility. S&ID will qualify the console at the system level.

Figure 5-60 displays the qualification test schedule for the IFTS and defines the test utilization of the hardware enumerated in Table 5-24.

The specific test levels to be applied in qualification testing are found in the tables of environmental test criteria in Section 4.2 and are a function of the zone location of the equipment in the spacecraft. The zone number is given in Table 5-24.

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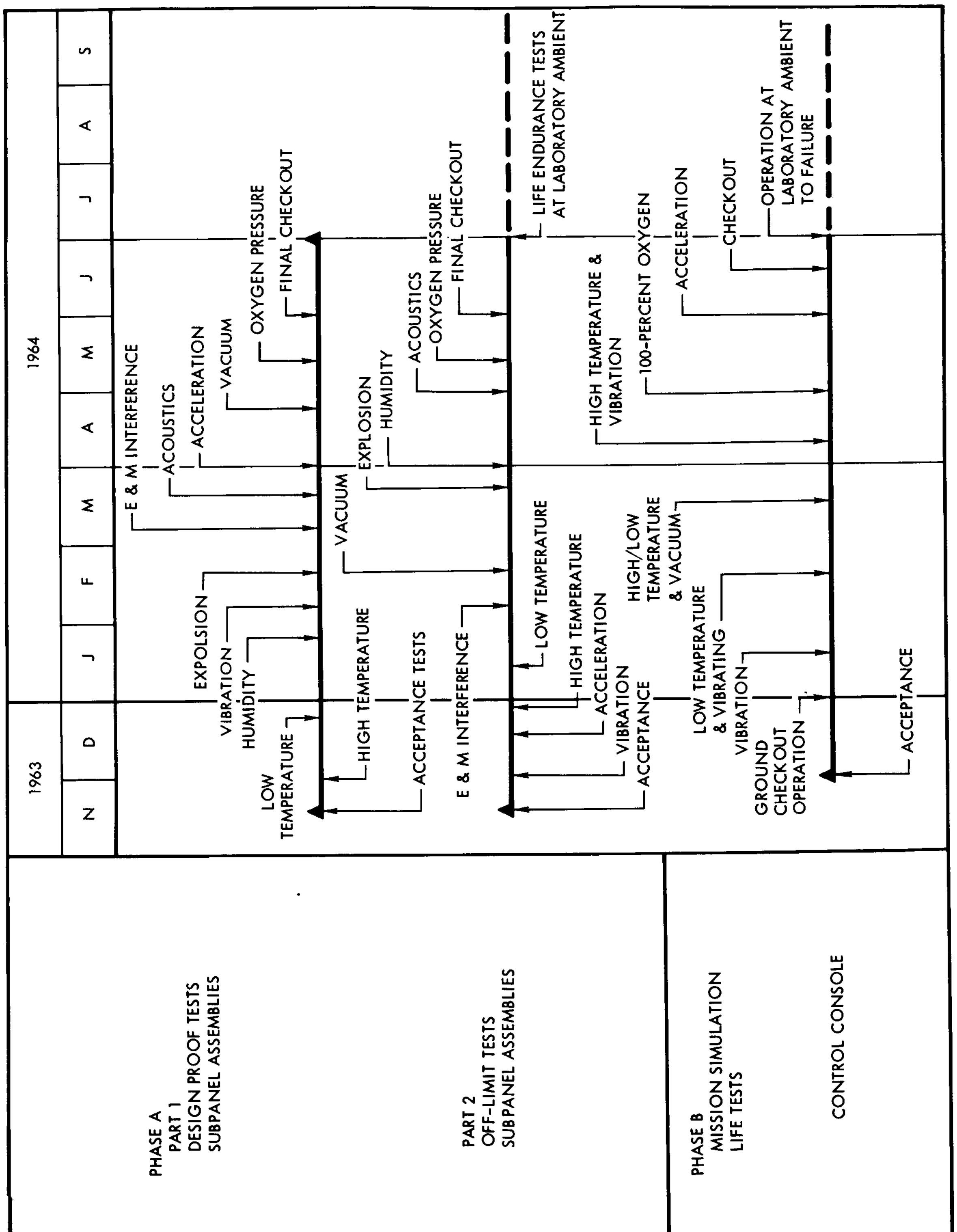

Figure 5-59. D & C Qualification Test Schedule

[redacted]

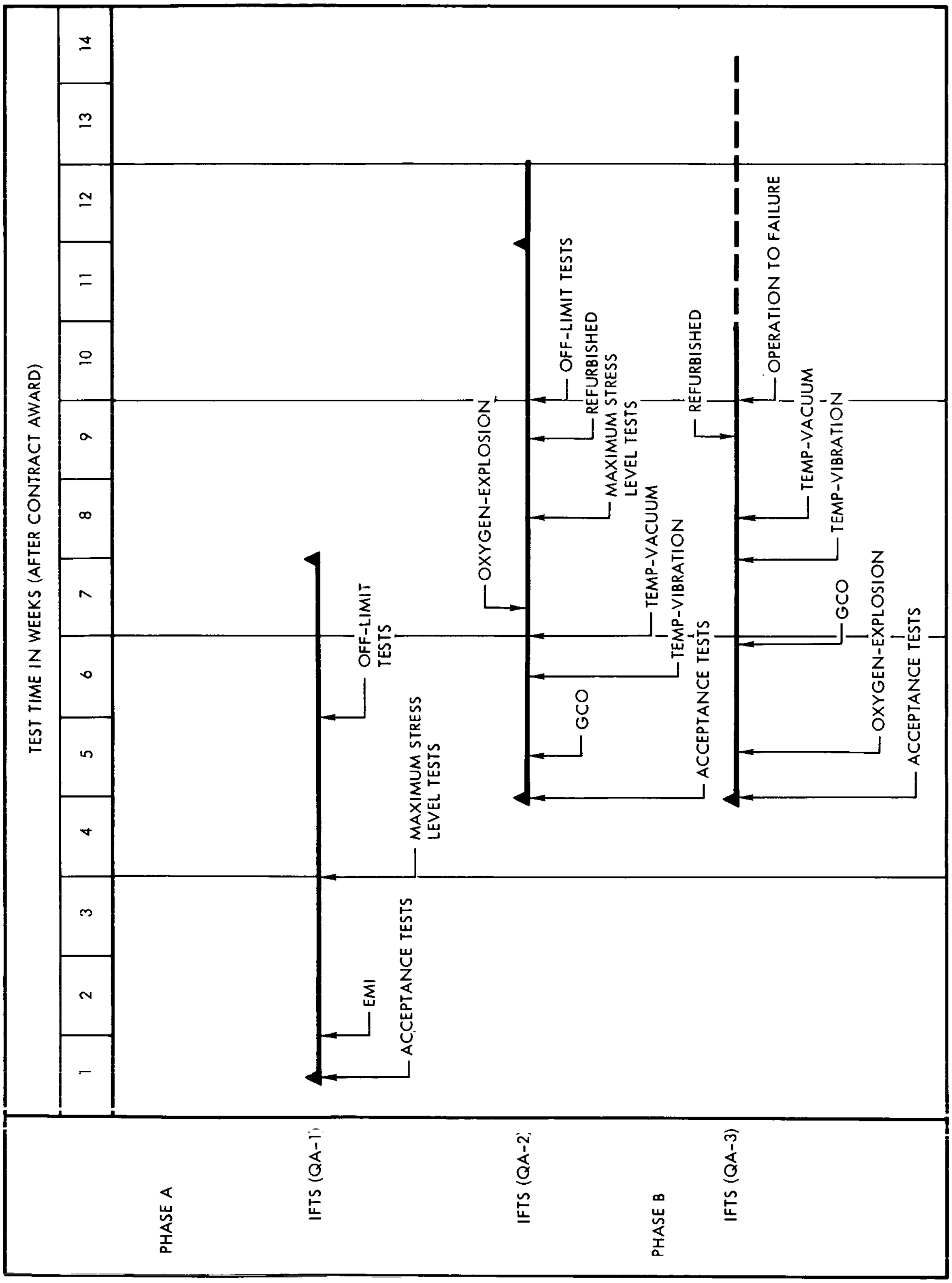

Figure 5-60. In-Flight Test System Qualification Test Schedule

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Table 5-24. ES Parameters for Qualification Testing

Subpanel Assemblies and Component Parts	S/C Zone	Criti-cality	Required Hardware	Test Phase Distribution		
				A-1	A-2	B
1. Barometric indicator	2	3	5	5	2	2
2. Entry monitor display*	2	2	5	4	2	2
2.1 Switch, toggle						
2.2 Indicator light						
3. Emergency detection system and gimbal drive control	2	2	5	4	2	2
3.1 Switch, toggle (4 each)						
3.2 Annunciator light (3 each)						
3.3 Meter, electrical						
4. FDAI (SCS)*						
5. Emergency detection system and sequencer display	2	1	7	5	2	1
5.1 Switch, push-type (3 each)						
5.2 Annunciator light (29 each)						
5.3 Timer, event, digital countout						
6. Attitude set and GPD*						
6.1 Switch, toggle						
6.2 Potentiometer (5 each)						
6.3 Attitude indicator (3 each—roll, pitch and yaw)						
6.4 Indicator, gimbal position (yaw and pitch)						

[redacted]

Table 5-24. ES Parameters for Qualification Testing (Cont)

Subpanel Assemblies and Component Parts	S/C Zone	Criti-cality	Required Hardware	Test Phase Distribution A-1	A-2	B
7. ΔV display*						
7.1 Indicator, ΔV remaining						
7.2 Switch toggle						
7.3 Indicator ΔV, set						
7.4 Indicator tail-off						
8. Spare panel						
9. SCS control and mode select*						
9.1 Switch, toggle (9 each)						
9.2 Switch, pushbutton (8 each)						
10. Master caution (left-hand)	2	3	3	2	2	1
10.1 Annunciator (19 each)						
11. Master caution (right-hand)	2	3	3	2	2	1
11.1 Annunciator (21 each)						
12. RCS, GMT, static port valve	2	2	5	3	2	1
12.1 Meter, electrical						
12.2 Annunciator						
12.3 Meter (2 each)						
12.4 Digital counter						
12.5 Switch, rotary						
12.6 GMT, display						
12.7 Switch, toggle						

[redacted]

Table 5-24. ES Parameters for Qualification Testing (Cont)

Sub-Panel Assemblies and Component Parts	S/C Zone	Criticality	Required Hardware	Test Phase Distribution A-1	A-2	B
13. Radiation, audio, cryogenic and ECS system	2	2	5	3	2	1
13.1 Switch, toggle (13 each)						
13.2 Potentiometer (4 each)						
13.3 Meter, electrical (7 each)						
14. G&N computer control*						
14.1 Annunciator (11 each)						
14.2 Potentiometer (2 each)						
14.3 Switch, pushbutton						
15. Reaction control system	2	2	5	3	2	1
15.1 Switch, toggle (18 each)						
15.2 Light, indicating (16 each)						
16. Crew safety control panel						
16.1 Switch, toggle (12 each)						
17. Oxygen warning						
17.1 Annunciator						
18. Fuel cells and electrical power system						
18.1 Switch, toggle (28 each)						

[redacted] [redacted]

Table 5-24. ES Parameters for Qualification Testing (Cont)

Sub-Panel Assemblies and Component Parts	S/C Zone	Criti-cality	Required Hardware	Test Phase Distribution		
				A-1	A-2	B
18.2 Meter, electrical (4 each)						
18.3 Switch, rotary (4 each)						
18.4 Light, indicating (14 each)						
19. Antenna control system						
19.1 Switch, toggle (5 each)						
19.2 Switch, rotary						
19.3 Indicator, position, roll and yaw						
19.4 Indicator, tune to max						
19.5 Annunciator (2 each)						
20. Command data-link and service propulsion system	2	2	5	3	2	1
20.1 Switch, toggle (30 each)						
20.2 Potentiometer						
20.3 Switch, rotary						
20.4 Meter, electrical (3 each)						
20.5 Switch, rotary (4 each)						
20.6 Indicator						
20.7 Display, digital counter						
20.8 Annunciator (4 each)						

[redacted]

Table 5-24. ES Parameters for Qualification Testing (Cont)

Sub-Panel Assemblies and Component Parts	S/C Zone	Criticality	Required Hardware	Test Phase Distribution A-1	A-2	B
21. Right-hand side console, bus, switching						
21.1 Switch, rotary (9 each)						
21.2 Switch, toggle						
22. Right-hand side console circuit breaker and bus switching	2	3	3	2	1	1
22.1 Circuit breaker switch (83 each)						
22.2 Switch, toggle (7 each)						
22.3 Switch, rotary (6 each)						
23. Right-hand side console audio and lighting control	2	2	5	3	2	1
23.1 Potentiometer (3 each)						
23.2 Switch, toggle (7 each)						
23.3 Rheostat						
24. Left-hand side console, mission sequence controls	2	2	5	3	2	1
24.1 Switch, toggle (8 each)						
25. Left-hand side console, circuit breaker and SCS power	2	2	5	3	2	1
25.1 Circuit breaker switch (50 each)						

[redacted]

Table 5-24. ES Parameters for Qualification Testing (Cont.)

Sub-Panel Assemblies and Component Parts	S/C Zone	Criti-cality	Required Hardware	Test Phase Distribution		
				A-1	A-2	B
25.2 Switch, rotary (4 each) 25.3 Switch, toggle 26. Left-hand side console, audio and lighting control 26.1 Rheostat 26.2 Switch, toggle (7 each) 26.3 Potentiometer (3 each)	2	3	3	2	1	1
SYSTEM LEVEL						
Control console in-flight test system	2	2	3	3	2	1

*Qualified by Vendor, Subcontractor or Associate Contractor

[redacted]

6.0 GROUND SUPPORT EQUIPMENT

6.1 SCOPE

This section is devoted to detailed descriptions of the qualification test program as applied to the various types and classes of ground support equipment required to support the Apollo Project and that will be the responsibility of S&ID. The program has many constraints which govern and limit the scope of the qualification program. The test planning has, therefore, been designed to work within these bounds.

The sample size has been limited, in most cases, to one item. In some cases, that item is also the end item to be delivered. Additional items will be required for certain mission essential classes where the contribution of the component to the mission is comparable with spacecraft systems requiring many more components for qualification.

There are many categories and classes of GSE required in support of Apollo and varying degrees of essentiality within these categories; each must have a program specifically tailored to the class and item. This document will cover only the basic categories.

In many cases, the time available for qualification testing is severely limited, allowing time for only a bare minimum of qualification; in some instances, qualification tests approach acceptance tests in scope. This is particularly true for items that are required for support of the early boilerplate programs.

Other constraints, such as facilities scheduling and varied environments at the points of application, have complicated the program planning. In spite of this, the program, presented herein, is considered a reasonable compromise.

6.2 GSE QUALIFICATION TEST LOGIC

The over-all concept of the GSE test program is similar to that used for the spacecraft equipment; however, the sample sizes are more severely limited. The sample sizes and the scope of the test program are a function of the particular contribution to the mission and the criticality class of the test item.

Qualification tests of appropriate parts, components, subassemblies, and higher levels of assembly will demonstrate that the design is inherently capable of meeting the performance and environmental requirements specified for the application. The tests will be performed at the most practical level of assembly. Qualification tests are divided into two phases: phase A, design proof, and phase B, life tests.

The tests are also categorized by the equipment groups. The groups being defined on the basis of ramifications in mission success and crew survival. See Figure 6-1 for the distribution by category and function. The test applications to those categories are covered in paragraph 6.5. The maximum qualification program, as applied to mission essential equipment (MEE, defined in paragraph 6.4.2), is composed of the following:

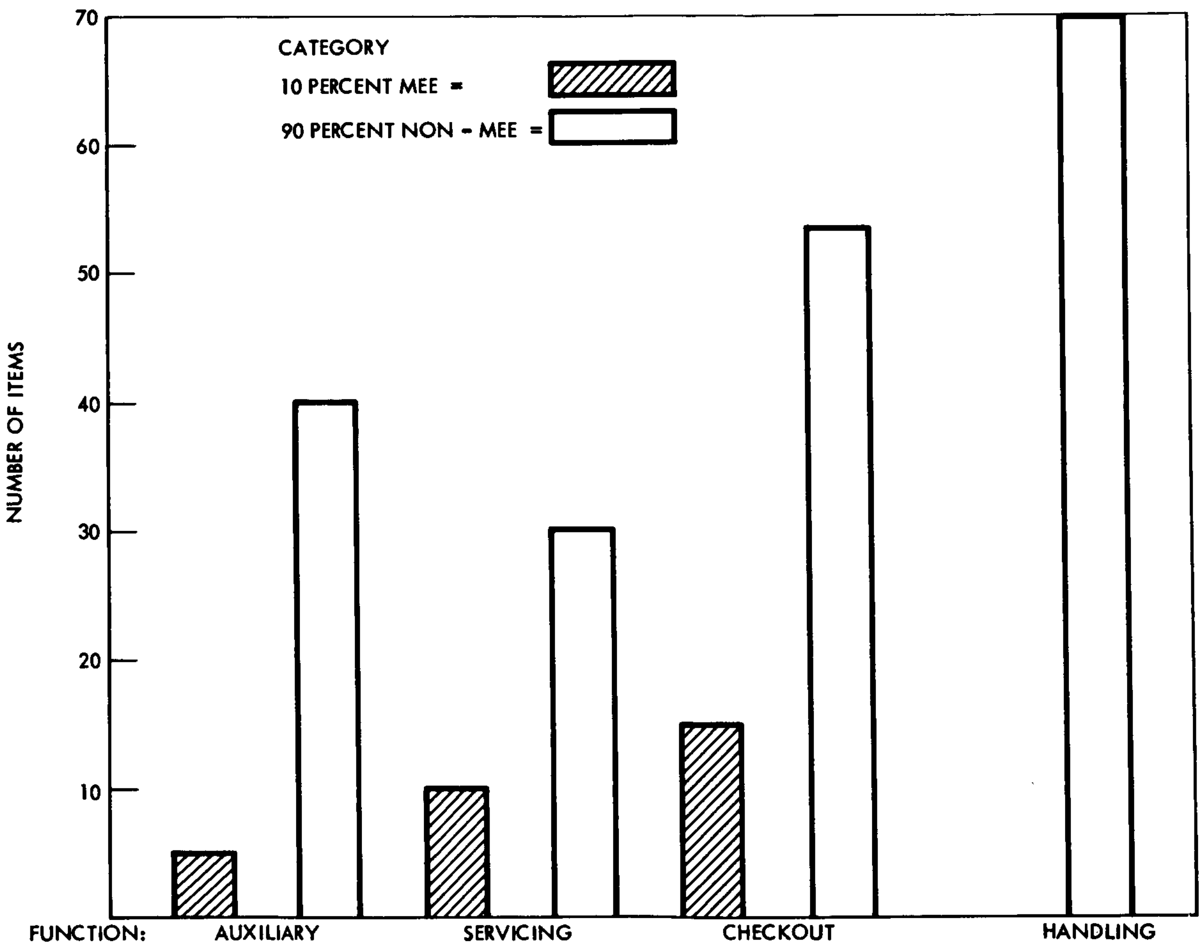

Figure 6-1. Distribution of Essential and Nonessential Ground Support Equipment

6.2.1 Phase A, Design Proof

Design proof testing includes:

1. Transportation and handling per MIL-E-4970A

2. Electromagnetic interference (EMI) per S&ID Specification MC 999-0002B

3. Off-limit tests, functional and nondestructive.

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6.2.2 Phase B, Life Tests

Life testing will be applied to mission essential units and certain other components requiring a long life because of their intended use. The tests will consist of exposure to natural environments as delineated herein. Each unit assigned to life tests will be required to complete a given number of simulated use-operation cycles without any failures. The number of cycles required for qualification will be limited to one, except for mission essential components that are determined as a function of criticality. The use of spares during the qualification cycle will not be permitted. Upon completion of the prescribed number of cycles for qualification, limited life items may be replaced, as required, and the equipment will then be subjected to additional cycles in order to make at least four cycles before the life test has been completed.

The number of cycles imposed on a piece of mission essential GSE to qualify it has been made a function of its contribution to the mission. This is defined by criticality class. The criticality classes for GSE were derived in much the same manner as those for the spacecraft, except that the definitions are based on the ramification of a GSE induced failure or a failure that was not detected. The remaining series of cycles are required to establish repeatability of the required characteristics, to demonstrate the life objective, and to provide maintainability assurance.

6.3 OPERATIONAL READINESS

6.3.1 Reliability Assessment

The reliability assessment program will be planned in accordance with apportioned program objectives. No additional hardware will be employed exclusively to assess reliability. The program includes specific ground rules for choice and application of data from all areas of testing. The actual assessment of reliability will be accomplished by an analytical treatment of data from tests designed to determine performance margins and life characteristics. Qualification testing will be monitored and controlled to assure maximum usefulness of the derived data for reliability assessment purposes.

6.3.2 Availability

It should be noted that reliability is not the best assessment figure of merit for GSE. Availability is a more applicable term and will be applied in the assessment program for Apollo GSE. Availability, as herein used, is analogous to operational readiness and is equal to the total operational in-service time divided by the sum of operational in-service time plus active repair time. Active repair time excludes logistic considerations. The

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measure of this active repair time is termed mean-time-to-repair (MTTR). A goal of 30 minutes has been set as the allowable active repair time.

6.4 GSE CATEGORIES

6.4.1 Concept

The GSE must be divided into basic categories in order to properly establish the scope of the qualification program. It, therefore, has been divided as a function of the contribution to mission success or crew survival, as follows:

1. Mission essential equipment (MEE)
   a. Checkout
   b. Auxiliary
   c. Servicing

2. Mission nonessential equipment (MNEE)
   a. Checkout
   b. Auxiliary
   c. Servicing
   d. Bench maintenance
   e. Handling

6.4.2 Definition of Mission Essential (MEE)

A piece of GSE is classified as mission essential if it is in a closed loop with the spacecraft just prior to or during countdown, so that it:

1. May fail to detect a malfunction
2. May introduce a malfunction into spacecraft equipment
3. May cause a complete countdown recycle

6.5 GSE ENVIRONMENTS

The GSE is to be subjected to two classes of environments as a function of equipment use and location. The environments to be applied are presented by equipment category in Table 6-1. The detailed values applied are given in the specific test plans in subsequent paragraphs. It should be noted that salt fog, sand and dust, and rain are only applied to mission essential equipment employed in unsheltered areas. As in spacecraft tests, it has been assumed, with NASA concurrence, that the GSE will receive special handling.

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Table 6-1. GSE Qualification Test Environments and Sequence

Test Environments	Mission-Essential		Mission-Nonessential	
	Sheltered Equipment	Unsheltered Equipment	Sheltered Equipment	Unsheltered Equipment
Function Checkout	1	1	1	1
High Temperature	4	4	4	4
Low Temperature	3	3	3	3
Low Pressure	2	2	2	2
Humidity	7*	7	7*	7*
Vibration	6	6	6	6
Shock	5	5	5	5
Salt Fog		8*		
Sand and Dust		9*		
Rain		10*		
Operating Temperature		11		8
Tilted Position	8*	12*	8*	9*
Electromagnetic Interference	9	13	9	10
Noise	10*	14*	10*	11*
Lifting	11*	15*	11*	12*
Dielectric	12	16	12*	13*
Insulation	13	17	13*	14*
Continuity	14	18	14*	15*
Life	15	19	15	16

*Where applicable

Note: The numbers indicate the suggested sequence or application

[redacted]

6.6 QUALIFICATION FOR MISSION ESSENTIAL EQUIPMENT (MEE)

6.6.1 Design Proof Tests

Design proof tests are generally conducted in accordance with MIL-E-4970A. They consist of three different steps: environmental, EMI, and functional off-limit tests.

6.6.1.1 Environmental Tests

General test conditions, tolerances, etc., will be in accordance with requirements of Section 3 of MIL-E-4970A, as follows:

6.6.1.1.1 Temperature. Temperature tests will be conducted to determine the effects of high and low temperatures on equipment during transportation, storage, and operation.

6.6.1.1.1.1 High Temperature (Nonoperating). Nonoperating high temperature tests will be performed according to Paragraph 4.1.2 of MIL-E-4970A, except that the test temperature will be reduced to +140 F because of the modified environments applied to Apollo.

6.6.1.1.1.2 Low Temperature (Nonoperating). The nonoperating low temperature test will be performed according to Paragraph 4.3.3 of MIL-E-4970A with the following modifications:

1. The chamber temperature will be reduced to minus 20 F because of the modified environments applied to Apollo GSE.

2. The operational check will be accomplished when the temperature of the item returns to the prevailing room temperature.

6.6.1.1.1.3 Operating Temperature Tests. The equipment will be placed within the test chamber and the internal temperature of the chamber will be raised to +120 F. The equipment will be operated while exposed to this temperature for a period of 42 hours. The chamber temperature will then be lowered to +30 F for another 42 hours of operation. While in the temperature chamber, the equipment operation will simulate the in-service requirements, and the performance during this test will conform to the requirements of the governing document.

6.6.1.1.2 Low Pressure. Low pressure tests will be conducted to determine the ability of the equipment to operate satisfactorily under reduced pressure conditions where higher altitude operation is deemed a potential problem and also to determine the ability of the equipment to withstand air transportation in a nonoperating condition.

6.6.1.1.2.1 Procedure. The low pressure tests will be performed according to Paragraph 4.4.1.1 of MIL-E-4970A, except that the internal absolute pressure will be reduced to 7.04 inches of mercury (corresponding to an altitude of 35,000 feet above sea level) instead of 3.42 inches of mercury. This is applied only as a nonoperating environment where air transit is involved.

6.6.1.1.3 Humidity Tests. Humidity tests are conducted to determine the resistance of equipment to the effects of exposure to a warm, highly humid atmosphere, such as is found in the beach areas at AMR.

6.6.1.1.3.1 Procedure. Humidity tests will be performed in accordance with Paragraph 4.3.1 of MIL-E-4970A as modified below:

1. The temperature for the first portion of each cycle will be 122 F.

2. Total test time for the humidity test will be 48 hours (2 cycles).

6.6.1.1.4 Vibration Tests. Vibration tests are conducted on equipment to determine the effects of vibrations encountered during transportation and handling.

6.6.1.1.4.1 Procedure. The vibration test procedure will be in accordance with Procedure VI of MIL-E-4970A, Paragraph 4.6.8. For items of less than 1000 pounds, Table III of Paragraph 4.6.8 will be used. For items over 1000 pounds, the Table 6-2 will apply:

Table 6-2. Vibration Tests for Items Over 1000 Pounds

Required Range CPS	Vibratory Double Amplitude or Peak Acceleration	Type of Test
2-20	± 1.04g	Resonance search only
20-52	0.029 inch	Cycling and at resonance
52-500	± 3.33g	Cycling and at resonance

6.6.1.1.5 Shock Tests. Shock tests will be conducted on equipment to determine the effects of shocks to be encountered during transportation and servicing.

6.6.1.1.5.1 Procedure. The shock tests will be conducted in accordance with Paragraph 4.12.2 and 4.12.6 of Specification MIL-E-4970A for

[redacted]

items of equipment weighing less than 1000 pounds. For items weighing over 1000 pounds, the tests will be the same, except an 18-g shock will be applied instead of the 20-g shock because of anticipated dampening effects.

6.6.1.1.6 Sand and Dust Tests. Sand and dust tests will be performed only on equipment that is sensitive to sand and dust and is expected to have an end use in an unsheltered area.

6.6.1.1.6.1 Procedure. The sand and dust test procedure will be in accordance with Procedure I of Paragraph 4.10.1 of MIL-E-4970A.

6.6.1.1.7 Salt Fog. Salt fog tests will be conducted only on equipment that may be sensitive to a salt atmosphere as encountered in ocean coastal areas and has an end use in an unsheltered area.

6.6.1.1.7.1 Procedure. Salt fog tests will be conducted in accordance with Paragraph 4.5.1 of MIL-E-4970A.

6.6.1.2 Electromagnetic Interference Test

Electromagnetic interference tests will be applied to relevant GSE components as delineated in S&ID Specification MC 999-0002B without modification.

6.6.1.3 Off-Limit Test

Off-Limit tests, as applied to Apollo GSE, are functional tests designed to verify that the design is resistant to critical failure modes. The specific application of this test will be determined by failure mode and effect analyses currently in process. Specific data will be contained in later revisions of this document. The tests will be applied only to mission essential equipment in GSE criticality Classes I and II, as defined in Paragraph 6.6.2.2. They will be designed to be nondestructive in nature.

6.6.2 Life Test

6.6.2.1 Concept

Life tests, as applied to mission essential equipment, are a series of tests designed to verify tolerance to critical parameter variations and total equipment performance as a function of accrued operating time. The tests are to be performed, using limited combined environments, when and if a failure mode analysis indicates potential critical failure modes in these environments, within the definition of Paragraph 6.4.2.

Life tests provide a second and equally necessary function, that of determining the availability (A) of the equipment and its average downtime or mean-time-to-repair (MTTR). This is necessary in order to establish that the equipment can, in case of a malfunction, still meet the mission objectives. It is anticipated that a reasonable level of reliability and maintainability data will be derived as a result of these tests.

Life tests, as herein described, are applied predominantly to mission essential equipment. A modified version, however, is applied to dynamic equipment in other GSE categories when usage demands. These are employed only to demonstrate a life requirement, compatible with value consideration and an operational readiness stature.

6.6.2.2 Criticality as applied to GSE

The time of exposure to the test cycle has been derived on a compatible basis with that of the spacecraft mission life tests (MLT) and is a function of service life and criticality. Since the spacecraft MLT time was based on the contribution of the equipment to mission success or crew survival, criticality for GSE is similarly defined. However, instead of a rigorous calculation of a specific criticality number, three criticality categories were established by qualitative definition.

The following is a list of criticality classes and the definitions of each:

Class I = May prejudice crew survival

Class II = May cause a pad or in-flight abort

Class III = May cause a countdown recycle (including limited holds)

(Within the definition of mission essential of Paragraph 6.4.2)

6.6.2.3 Application

6.6.2.3.1 Test Design. The life test is designed to accommodate all classes of criticality. However, qualification is determined on the basis of the specific class. Reference is made to Figure 6-2, where a time line diagram is given for the life test. Each component subjected to the life test is required to complete four cycles of operation consisting of a simulated preparation (prior-to-launch) phase and a mission (pad operations) phase. These tests may be terminated at the life requirement if it is less than the time equivalence of four full cycles. Some refurbishment will be allowed and, in fact, is anticipated because of the small sample sizes. Qualification of the component is contingent on passing Cycle 1 for Class III criticality.

[redacted]

Cycle 2 for Class II criticality, and Cycle 3 for Class I criticality. Stipulated failure rates, maintenance time constraints, and availability must not be exceeded. These are not based on statistical confidence but merely upon a simple calculation employing formulae which yield no statistical confidence but do provide some indication of equipment capability.

$$\text{Mean time between failure (MTBF)} = \frac{\text{Total test time}}{\text{Number of failures}} = \begin{array}{l}\text{500 hours MEE} \\ \text{300 hours MNEE}\end{array}$$

$$\text{Maintenance time constraint (MTTR)} = \frac{\text{Downtime (active)}}{\text{Number of failures}} = \text{30 minutes}$$

$$\text{Availability (A)} = \frac{\text{MTBF}}{\text{MTBF + MTTR}} = \text{0.999 (for MEE)}$$

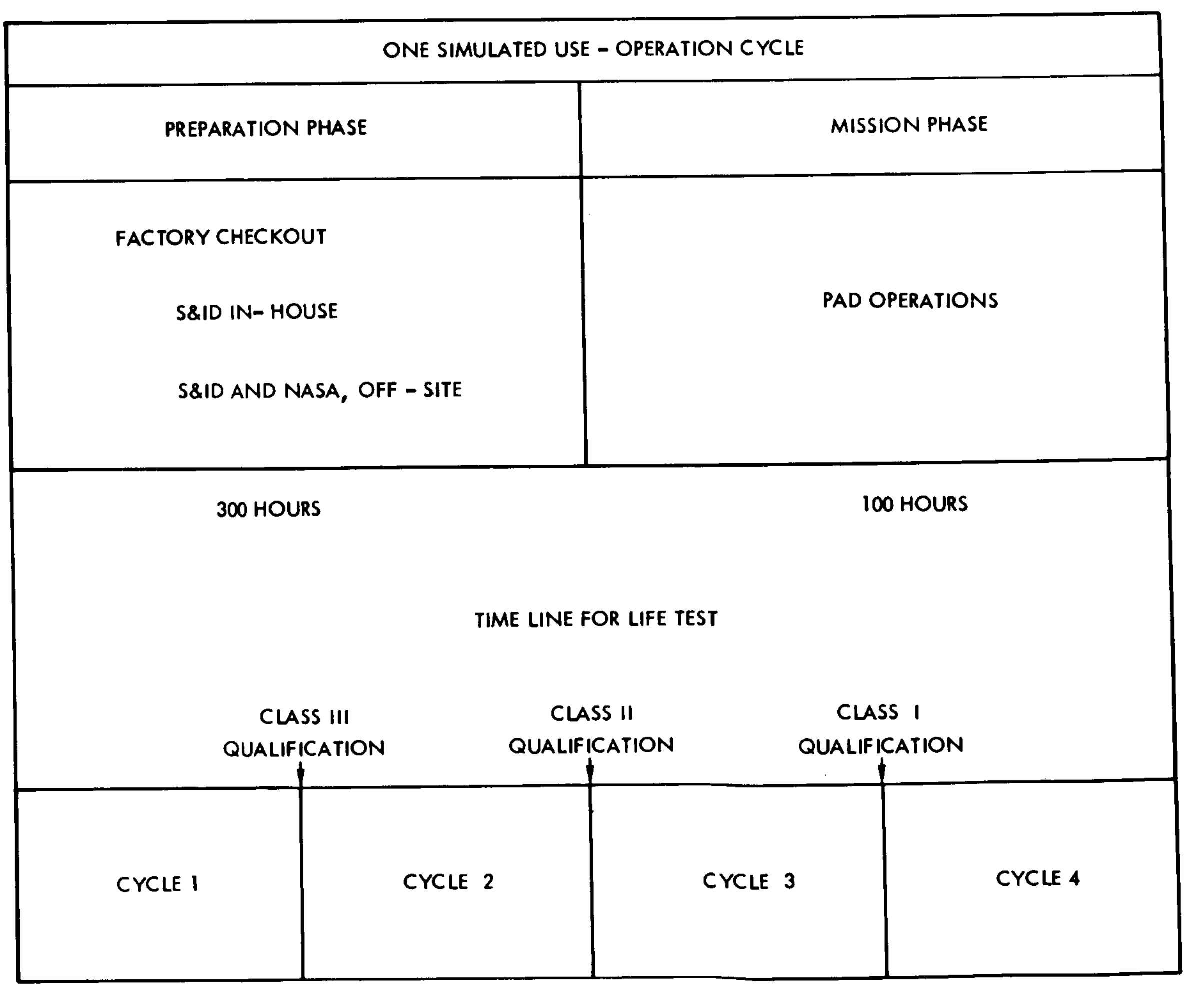

Figure 6-2. Life Test Application

6.6.2.3.2 Use — Operation Cycle Time. The time imposed for one-cycle use operation is derived from the following calculation. This is based on past experience, extrapolated from what is defined and anticipated for Apollo. These are average times and are not considered rigorous at this time.

Factory checkout and acceptance	65 hours
S&ID Downey use	110 hours
On-site (prior to launch operations)	125 hours
Pad use (includes three countdown recycles)*	100 hours
	Total 400 hours

*Applied only to MEE

6.7 QUALIFICATION TESTS FOR MISSION NONESSENTIAL EQUIPMENT (Exclusive of Handling Equipment)

6.7.1 Design Proof Tests (MNEE)

The design proof tests, as applied to the mission nonessential equipment, exclusive of handling equipment, will be based also on MIL-E-4970A, except that the EMI tests will be as delineated in S&ID Specification MC 999-0002B. The specific series of environments to apply to a given category of nonessential GSE is delineated in Table 6-1.

6.7.2 Life Tests (MNEE)

Life tests, as applied to the mission nonessential GSE, are a modified version of those specified for the MEE, as delineated in Paragraph 6.6.2. The basic differences are that they will always be conducted under local ambient conditions and that qualification will be based on the specified life of the equipment. The test will not be applied to all equipments in this category but will be limited to those dynamic type equipments deemed to be critical enough to require life demonstration.

6.8 QUALIFICATION TESTS FOR HANDLING-TYPE EQUIPMENT

The qualification tests planned for this type of equipment are limited to an expanded acceptance-type test since no items are set apart exclusively for qualification. All components in this category will be subjected to non-detrimental tests, as delineated in Paragraph 6.8.1. The specific application of these tests varies considerably between components and will be covered in the subsequent revision of this volume.

[redacted] [redacted]

6.8.1 Test Types

1. Detailed examination of product

2. Special Nondestructive analysis, such as infrared, magnaflux, and X-ray

3. Proof load tests

Specific information on the nature and application of these tests is given in Volume IV, Acceptance Test Plan, of this series.

[redacted]

[redacted] [redacted]

7.0 INTEGRATED SYSTEMS QUALIFICATION*

7.1 SCOPE

This section will be confined to qualification and reliability program requirements and objectives to be accomplished prior to and in support of any and all phases of integrated systems testing. In the logical evolution of a fully qualified Apollo spacecraft, test objectives must be established that assure component and subsystem qualification or flight worthiness for each boilerplate or airframe test. These test objectives are best realized by preflight analysis of vehicle readiness and establishment of success criteria termed "interim qualification requirements" based on the mission objectives, which in turn form the control measures to be satisfied before a given flight should be attempted.

7.1.1 Purpose

A twofold purpose exists for the qualification requirements of the integrated systems tests. In chronological order, these are to assess the flight readiness of the test vehicle and its components and to predict the probability of success of the mission relative to its objectives. The assessment of flight readiness and the prediction relevant to test success serve to optimize the extent and interaction of each test objective in relation to the over-all qualification program.

7.1.2 Method

A flight/qualification flow chart has been developed to show the philosophy and implementation of integrated systems testing. Interim qualification requirements are delineated in tabular form for each flight vehicle in ascending order of severity as the mission objectives become increasingly more complex and extensive. The intent is to always ensure a reasonable probability of accomplishing the mission objectives.

7.2 FLIGHT/QUALIFICATION FLOW CHART

The flight/qualification flow chart presents a composite picture of the integrated systems test program. The boilerplate test program (Figure 7-1),

*Entire section reissued.

[redacted]

with its corresponding on-board systems, is separated from the airframe program (Figure 7-2). This should not, however, detract from the continuity of flow which exists between the two.

7.2.1 Flight Test Vehicles

The flight articles are blocks arranged horizontally to correspond to the test site with their flight dates represented by the right border. The block length has no scheduling significance.

7.2.2 Nonflyable Vehicles

Nonflyable articles are the cross-hatched areas drawn to correspond to their respective test schedules. These articles include the house spacecrafts, B-14, AFRM 006, the propulsion test spacecraft, AFRM 001, and the environmental proof airframe, AFRM 008, from which the test data will be utilized to support qualification of flight articles for all phases of the test program.

7.2.3 Flow Chart Utilization

Figure 7-1 depicts the integrated system test planning flow and the logical evolution of the ground qualification program as they parallel and augment the flight test program. The intent is to ensure a well-balanced test program, accomplishing first those tests that can be performed under simulated conditions on the ground and then augmenting and verifying results by actual flights. The qualification flow is shown by lines accompanied by a statement as to anticipated qualification status at key points. Some of these are between the particular flight articles; and between the articles and the integrated systems tests, such as propulsion tests and environmental tests conducted at other facilities. Individual systems qualification and development test status is shown relative to the launch dates. In the next revision, the explicit installation dates for the system in each test article will be shown as vertical arrows on the system test bar at the appropriate date. This will not only indicate in which article each system is to be installed, but will also mark the stage of development or qualification testing each system has undergone prior to its installation.

7.2.4 Sample Application

To illustrate the utilization of the flight/qualification flow charts, Figures 7-1 and 7-2, B-12 is chosen as an example. The mission assignment for B-12 is the Max-q LES verification. From the flow of the boilerplate test program (Figure 7-1), it may be seen that B-6, whose mission assignment is pad abort, precedes B-12. Also, the LES tower and motors, the parachutes, and the solid-state abort sequencer are not qualified

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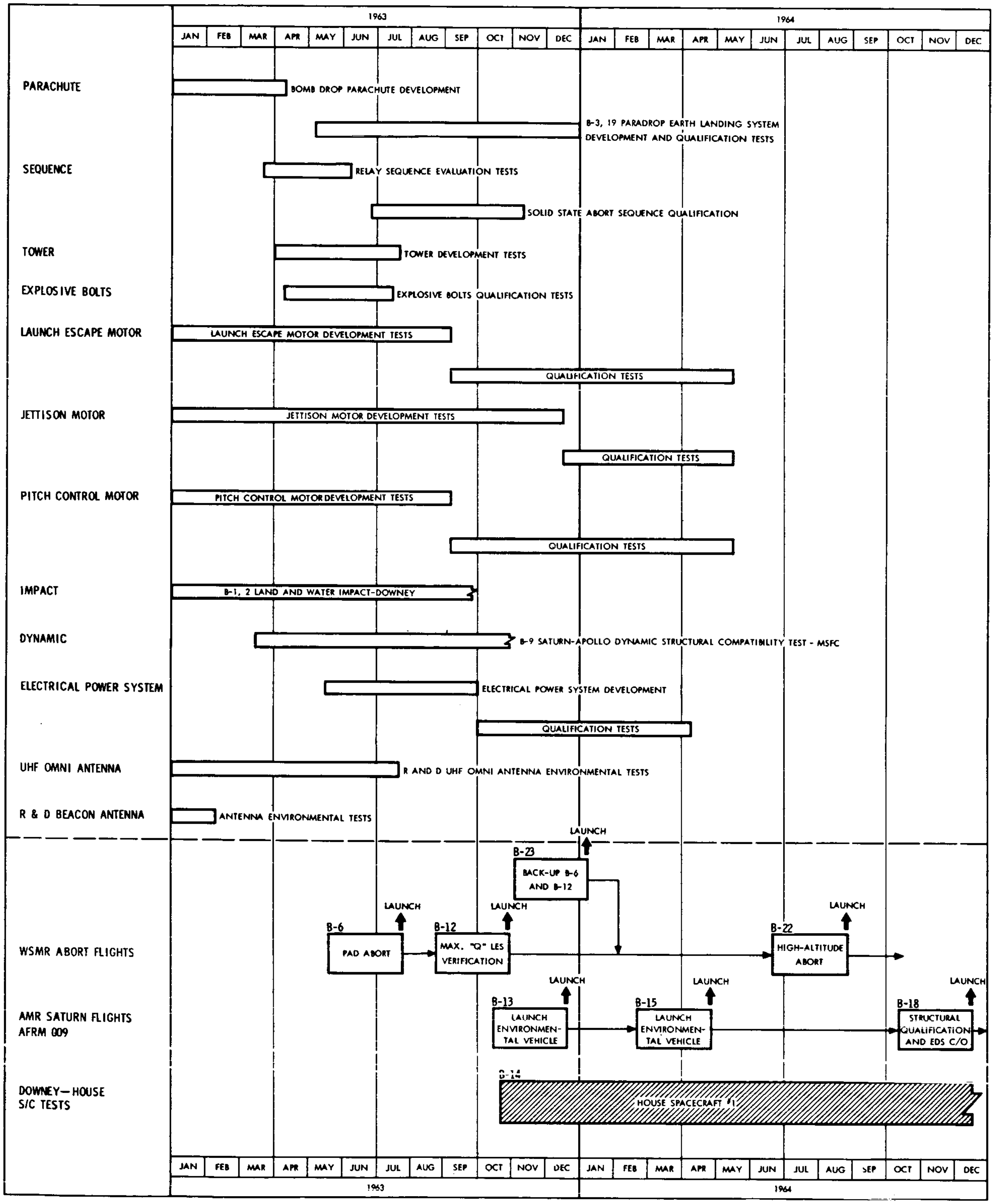

Figure 7-1. Boilerplate Flight/Qualification Flow Chart

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[redacted]

at the time of installation on board B-12. The explosive bolts, however, are qualified. The qualification requirements for B-12, therefore, will consist of two groups. The first group will be components and systems that are ground-qualified prior to flight and may include partial flight qualification, such as explosive bolts on B-6. The second group will be those components and systems that are not ground-qualified at the time of installation on the specific test vehicle. Special qualification requirements, to be known as "interim qualification requirements," are generated to encompass the latest status of development for the latter group.

These special tests are usually extracted from the applicable procurement specifications or Apollo test requirements, each of which has specified the test required to qualify the systems. A representative selection is made based upon the types of environments the flight will encounter. The sequence of testing duplicates the events as scheduled for the limited mission.

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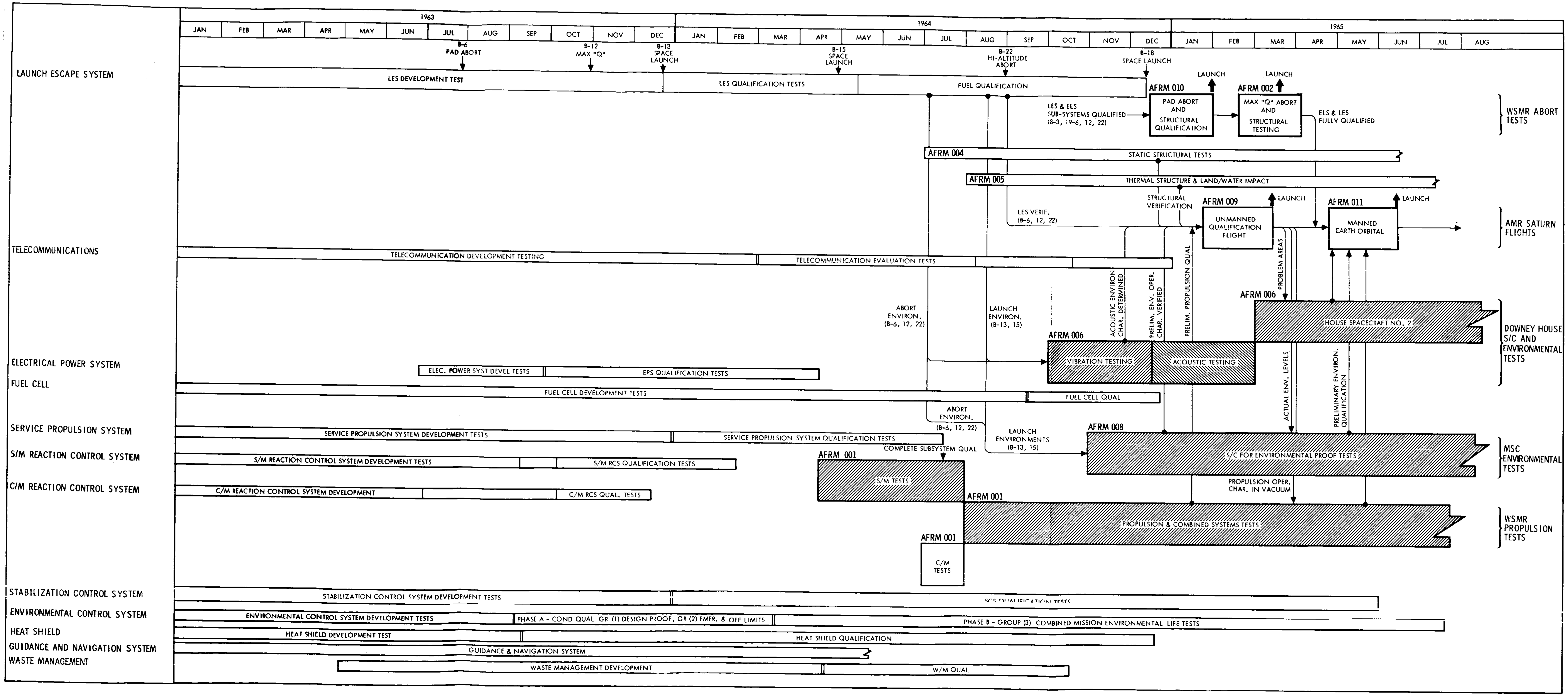

Figure 7-2. Airframe Flight/Qualification Flow Chart

[redacted]

NOMENCLATURE

A listing of the special nomenclature and abbreviations used in this report is presented. These listings may appear interchangeably with their full definitions in the text. Accepted abbreviations for units of physical measurement such as volts, ohms, etc. are not included.

Abbreviation	Definition
AEDC	Arnold Engineering Development Center
AFMTC	Air Force Missile Test Center
AFRM	Airframe
AGAP	Attitude gyro and accelerometer package
AGC	Aerojet-General Corporation
AGC	Automatic gain control
AGC	Apollo guidance computer
AGREE	Advisory group on reliability of electronic equipment
AMR	Atlantic Missile Range
ATO	Apollo Test and Operations
BAL	Balance
BCD	Binary coded decimal
B-#	Boilerplate - # (with specific number)
B/M	Bench maintenance
BMAG	Body-mounted attitude gyros
BOD	Beneficial occupancy date
B/P	Boilerplate

[redacted]

Abbreviation	Definition
CCMTA	Cape Canaveral Missile Test Annex
C&D	Controls and displays
CDS	Communications and data subsystems
CDU	Coupling display unit
CG	Center of gravity
C&IS	Communications and instrumentation system
C/M	Command module
C/O	Checkout
C-O	Crew operated
CP	Control panel
CTU	Central timing unit
CVR	Change verification record
DEA	Display electronic assemblies
DF	Direction finding
DOD	Department of Defense
DOF	Degrees of freedom
DP	Design proof
DPT	Design proof test
DSIF	Deep Space Instrumentation Facility

[redacted] [redacted]

Abbreviation	Definition
DVD	Differential velocity display
EBW	Exploding bridge wire
ECA	Electronic control assembly
ECS	Environmental control system
EDL	Engineering Development Laboratory (S&ID Dny)
ELS	Earth landing system
EMI	Electromagnetic interference
EPS	Electrical power system
ET	Escape tower
FDAI	Flight director attitude indicator
FM	Frequency modulation
FAX	Facsimile transmission
G, g	Acceleration of gravity
G&NS	Guidance and navigation system
GOSS	Ground operational support system
GP	General purpose
GPI	Gimbal position indicator
GSE	Ground support equipment
HAA	High-altitude abort
HBW	Hot bridge wire
H/A	Hazardous area

[redacted]

[redacted]

Abbreviation	Definition
IFT&M	In-flight Test and Maintenance
IMCC	Integrated Mission Control Center
IMU	Inertial measurements unit
INT	Interior
INST REF	Instrument reference
IRIG	Inter-range instrumentation group
L-	Time before launch (days)
LAD	Los Angeles Division (or NAA)
L/D	Length-diameter ratio
LEM	Lunar excursion module
LEV	Launch escape vehicle
LES	Launch escape system
LJII	Little Joe II
LOR	Lunar orbit rendezvous
LSS	Life support system
L/V	Launch vehicle
MDSS	Mission data support system
M-#	Mockup - # (with specific number)
Max q	Maximum dynamic pressure
MEE	Mission essential equipment

[redacted]

[redacted] [redacted]

Abbreviation	Definition
MIT	Massachusetts Institute of Technology
MLT	Mission life test
MNEE	Mission non-essential equipment
MSC	Manned Spacecraft Center (NASA, Houston, Texas)
MTBF	Mean time before failure
MTTR	Mean time to repair
NAA	North American Aviation
NASA	National Aeronautics and Space Administration
O/F	Oxidizer-to-fuel ratio
OPS	Operations
OTP	Operational test procedure
PA	Power amplifier
PACE-S/C	Prelaunch automatic checkout equipment - spacecraft
PAM	Pulse amplitude modulation
PCM	Pulse coded modulation
PCM	Pulse code modulator
PDM	Pulse duration modulation
P_f	Probability of failure
P_{f_p}	Probability of performance failure
P_{f_s}	Probability of stress failure
PFRT	Preliminary Flight Rating Test
POD	Prelaunch Operations Division (NASA)

[redacted]

[redacted]

Abbreviation	Definition
P_s	Probability of success
PSA	Power and servo assembly
PTT	Push to talk
PUCS	Propellant utilization control system
P&WA	Pratt & Whitney Aircraft
R/B	Radar beacon
R/C	Radio command
RCC	Range control center
RCS	Reaction control system
R&D	Research and development
REG	Regulator
RF	Radio frequency
RFI	Radio frequency interference
RFWAR	Requirements for work and resources
RGP	Rate gyro package
R&Z	Range and zero
SA	Saturn Apollo
SCAT	Space communications and tracking
SCD	Specification control drawing
SCIP	Self-contained instrumentation package
SCO	Subcarrier oscillator
SCR	Silicon-controlled rectifier

Abbreviation	Definition
SCT	Scanning telescope
S/C	Spacecraft
S/C_a	Spacecraft adapter
SCS	Stabilization and control system
SEP	Space electronic package
S&ID	Space and Information Systems Division (of NAA)
S/M	Service module
SOL	Solenoid
SOFAR	Sound fixing and ranging
SPS	Service propulsion system
STU	Systems test units
SXT	Sextant
T-	Time before launch
T-O	Time of launch
T+	Time after launch
T/M	Telemeter
TP	Test point
TPA	Test preparation area
TPS	Thermal protection system
T/R	Transmitter receiver (combination in two packages)
TWT	Traveling wave tube
TWX	Teletype wire transmission
UDMH	Unsymmetrical dimethyl hydrazine

[redacted]

Abbreviation	Definition
VCO	Voltage controlled oscillator
VLF	Vertical launch facility
VSWR	Voltage standing wave ratio
WSMR	White Sands Missile Range
XCVR	Transceiver (transmitter receiver in one package)

[redacted]

[redacted]

DEFINITION OF TERMS*

ABLATE	To carry away; to remove by cutting or erosion, melting or evaporation. To undergo ablation; to become melted or vaporized and removed at a very high temperature.
ABORT	An uncompleted missile flight or an uncompleted hold-down test resulting from a failure of equipment or of a system other than the one undergoing test. In a tactical operation (simulated or real) a missile failure either on the ground or in flight; a missile that fails to complete a programmed flight.
ADAPTER	Flange or extension of a vehicle stage or section providing a means of fitting another stage or section to it.
AEROBALLISTICS	Term derived from aerodynamics and ballistics, dealing primarily with the motion of bodies whose flight path is determined by applying the principle of both sciences to different portions of the path.
AEROTHERMODYNAMIC BORDER	Area above an altitude of about 100 miles where the atmosphere becomes so rarefied that there is no longer any significant heat-generating air friction on the skin of vehicles.
AIRFRAME	Assembled structural and aerodynamic components of an aircraft or missile.
ALBEDO	The ratio of light reflecting from an unpolished surface to the light falling upon it. Term is used in reference to light reflected from the moon or planets.
AMBIENT CONDITIONS	Environmental conditions such as pressure or temperature; naturally existing conditions.
ANTHROPOMORPHIC	Human-like; related to or designed for the human body.

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APOLLO

NASA designation for follow-up manned space-flight program to Project Mercury manned orbital mission. Apollo spacecraft is to be suitable for manned earth-orbiting laboratory, manned circumlunar flight, manned lunar landing, and return.

ATLANTIC MISSILE RANGE (AMR)

A 5000- to 6000-mile instrumented range for testing ballistic and guided missiles located between Cape Canaveral, Florida, and a point beyond Ascension Auxiliary AFB, near the middle of the South Atlantic.

ATTITUDE

Orientation of an air vehicle as determined by the inclination of its axis to a frame of reference, usually the earth.

ATTITUDE JETS

Sometimes called steering jets, attitude-control jets or roll, pitch, and yaw jets; fixed or movable gas nozzles on a rocket, missile, or satellite operated continuously or intermittently to change attitude or position.

AXIS, AXES

Reference axes in the Apollo spacecraft are as follows:

- **X-axis** — The X-axis is parallel to the nominal launch axis and is positive in the direction of initial flight.
- **Y-axis** — The Y-axis is normal to the X-axis and is positive to the right of a crewman when the crewman is in his seat facing toward positive "X."
- **Z-axis** — The Z-axis is normal to both the X- and Y-axes and is positive in the direction of the crewman's feet when he is in his seat.

BACKUP

Designed to immediately follow an earlier space system or a project to complement the latter or take advantage of new techniques and processes; a system that can replace a failed system.

[redacted]

BENCH MAINTENANCE EQUIPMENT	Equipment supporting component and subsystem testing; facilities capable of isolating, defining, and providing remedial action to malfunctions.
BIOMEDICINE	Combined discipline of biology and medicine for analysis of human tolerances to and protection against environmental variances.
BOILERPLATE	Simulated module for predevelopmental and developmental tests leading to the design of the spacecraft module.
BOOSTER ENGINE	An engine, especially a booster rocket, that adds thrust to the thrust of the sustainer engine, or provides propulsion for a special phase of flight.
BREADBOARD MODEL	An assembly of preliminary circuits and parts to prove the feasibility of a device, circuits, equipment, system or principle in rough or breadboard form without regard to the eventual over-all design or form of parts.
BRIDGE WIRE	The ignition resistor filament. The bridge wire heats the primary explosive to initiate the explosion.
CELESTIAL GUIDANCE	Mechanically or electrically recorded navigational tables, computers, and other instruments and devices that sight stars, calculate position, direct, and control the spacecraft.
CENTRIFUGE	A large motor-driven apparatus with a long rotating arm at the end of which human and animal subjects or equipment can be revolved at various speeds to simulate accelerations encountered in high-performance vehicles.
CHECKOUT	A sequence of operational and calibrational tests needed to determine the condition and status of a required operation or function.

[redacted]

COASTING FLIGHT	The flight of a rocket or other vehicle between burnout or thrust cutoff of one stage and ignition of another, or between final burnout and summit altitude or maximum range. Also the unpowered portion of an interplanetary flight.
COMMAND MODULE	Personnel and control vehicle in the Apollo spacecraft configuration containing command and communication facilities and crew provisions.
COMPATIBILITY	The quality that permits an item to function in harmony with other equipment and fulfill all design objectives.
CONFIGURATION	The physical nature of an item; the physical arrangement of components which comprise a spacecraft and its dimensions.
CONSOLE	Master instrument panel from which rocket and missile launchings and test are controlled; a group of controls, indicators, and similar electrical or mechanical equipment that is used to monitor readiness of and/or control specific functions such as missile checkout, countdown and launch operations.
COUNTDOWN	Series of numbered events and checks that take place from the start of rocket-launching operations until the rocket lifts off the launch stand.
CRYOGENIC FUEL	A rocket fuel that either in itself is kept at very low temperatures or combines with an oxidizer kept at very low temperatures.
CUTOFF	The shutting off of a liquid or solid-propellant combustion process of a rocket engine, causing a rapid drop toward zero thrust (intentional command action).
DEEP SPACE INSTRUMENTATION FACILITY (DSIF)	Communication equipment capable of contacting and tracking spacecraft beyond normal ranges. DSIF facilities are located at Woomera, Australia; Johannesburg, South Africa; and Goldstone, California

[redacted]

DEMONSTRATE	Denotes the occurrence of an action or an event during a test. The accomplishment of this type of objective requires a qualitative answer. The answer will be derived through the relation of this action or event to some other known information or occurrence. This category of objectives implies a minimum of system instrumentation and/or that information be obtained external to the test vehicle.
DESIGN VERIFICATION TEST	Basic development test used to determine the adequacy of the design over the anticipated operating conditions. A design verification test is always conducted by Engineering and on a breadboard level.
DETERMINE	Denotes the measuring of performance of any unit or system. This category implies the quantitative investigation of over-all operation, which includes, generally, instrumentation for measuring basic inputs and outputs of the unit or system. The information obtained should indicate to what extent the system is operating as designed. The instrumentation should allow performance deficiencies to be isolated to either the system or the system inputs.
DRAG	The resistance of a body to motion in a medium such as air.
DRY WEIGHT	Weight of a rocket vehicle without its fuel and usually without payload.
EARTH LANDING SUBSYSTEM (ELS)	Acceleration-decreasing equipment for return to the earth's surface after atmospheric reentry; may consist of a parachute system, a flexible aerodynamic glider configuration, or both.
ENGINEERING DEVELOPMENT PART	A part or unit to be employed in a breadboard design.
ENVIRONMENTAL CONTROL SUBSYSTEM (ECS)	The components controlling crew conditions in the spacecraft; governing factors of atmosphere, pressure, and temperature; and providing support for spacesuit conditions in event of cabin decompression or extra vehicular operations.

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ESTABLISH — Denotes gathering information for the development of ground procedures and operating techniques. Objectives in this category are not necessarily dependent on analytic studies.

EVALUATE — Denotes the measuring of performance of any unit or system, as well as the performance and/or interaction of its sections or subsystems that are under investigation. The accomplishment of objectives of this type requires quantitative data on the performance of both the unit or system, and its sections or subsystems will be analyzed for their contribution toward performance of the unit or system. This category will provide the most detailed information of any of these categories.

FALLAWAY SECTION — Any section of a rocket vehicle that is cast off and falls away from the vehicle during flight, especially such a section that falls back to earth.

FIRST MOTION — First indication of motion of the missile or test vehicle from its launcher. Synonymous with "takeoff" for vertically launched missiles.

FLIGHT READINESS FIRINGS (FRF) — A missile system test consisting of the complete firing of the liquid-propellant engines of a rocket missile while it is restrained in its launching stand to verify the readiness of the missile for a flight test or mission.

FREE-FLIGHT TRAJECTORY (Free Fall Ellipse) — That part of a ballistic missile's trajectory that begins with thrust cutoff and ends at reentry.

FUEL CELL — A source of electrical power analogous to a common electrical cell with the reactants continually replenished from an external supply.

GAMMA RADIATION — Electromagnetic radiation having a high degree of penetration similar to X-rays originating from the nucleus.

[redacted]

GEMINI — NASA follow-up program to Mercury; a two-man spacecraft to demonstrate rendezvous and docking techniques, longer orbital flights (to 14 days), controlled reentry, and landing.

GIMBALED MOTOR — A rocket motor mounted on a gimbal, i. e., on a contrivance having two mutually perpendicular axes of rotation so as to correct pitching and yawing.

GODDARD SPACE FLIGHT CENTER (GSFC) — NASA research center at Greenbelt, Maryland, named for Robert H. Goddard, American rocket pioneer.

GOLDSTONE TRACKING FACILITY — A deep space instrumentation facility located at Army's Camp Irvin, Barstow, California, using a radiotelescope and operated for NASA by Jet Propulsion Laboratory (JPL).

GO NO-GO — A missile launch controlled at the end of the countdown as to permit an instantaneous change in decision on whether or not to launch.

GROUND OPERATIONAL SUPPORT SYSTEM (GOSS) — Network of tracking stations, fixed and mobile, air, ground, and seaborne to communicate with, track, and telemeter spacecraft and satellites.

GROUND SUPPORT EQUIPMENT (GSE) — All ground equipment that is part of the complete spacecraft system and that must be furnished to ensure its support. All implements or devices required to maintain the functional operational status of the spacecraft are included. In the Apollo program, bench maintenance equipment, combined system test unit, and prelaunch automatic checkout equipment comprise the GSE.

GUIDANCE — (1) The process of intelligent gathering and maneuvering required by a missile, probe, or space ship to reach a specified destination. (2) General term includes entire scheme: sensing devices, computers, and servo systems.

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HEAT SHIELD	An ablative protective covering to ensure spacecraft and crew survival through the hypertemperatures of atmospheric reentry.
HEAT SINK	A device that absorbs heat energy.
HOT TEST	Propulsion system test conducted by actually firing the propellants. A hot test may be live, static, or conducted in a confined place.
INERTIAL GUIDANCE	An onboard guidance system for space and satellite vehicles where gyros, accelerometers, and possibly a gyro-stabilized platform satisfy guidance requirements without use of any ground-located components. The system is entirely automatic, following predetermined trajectory.
INFRARED LIGHT	Light in which the rays lie just below the red end of the visible spectrum.
INITIATOR	A primary explosive mixture used as a primer, detonator for caps which initiates the explosion of blasting propellant, bursting explosives at the desired moment.
INTERIM QUALIFIED	A term used to describe the status of a component or system scheduled for use in early flights wherein the basically essential qualification tests have been successfully completed as related to the specific flight objectives.
INVERTER	A converter to a-c power from a d-c source.
JET STEERING	The use of fixed or movable gas jets on a space weapon, ballistic missile, or sounding rocket for thrust vector control to steer it along a desired trajectory, during both propelled flight and after thrust cutoff.
KELVIN SCALE (K)	A temperature scale that used Centigrade degrees as gradients and absolute zero for zero. Zero Kelvin equals -460 F or -273 C.

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LANGLEY RESEARCH CENTER

NASA installation in Hampton, Virginia, responsible for technical research in development and improvement in both atmospheric and space flight.

LAUNCH WINDOW

The allowable limits of launch time that will allow a spacecraft to achieve successful injection into programmed flight path.

LAUNCH ESCAPE SUBSYSTEM (LES)

The components for command module recovery in case of mission abort after launch and prior to orbit. The system consists of the launch escape motor, the launch escape tower, and the tower jettison motor.

LIQUID HYDROGEN

Liquid rocket fuel that develops a specific impulse, when oxidized by liquid oxygen, ranging between 317 and 364 seconds depending upon the mixture ratio.

LIQUID OXYGEN

Oxygen supercooled and kept under pressure so that its physical state is liquid. Used as an oxidizer in a liquid-fuel rocket.

LITTLE JOE (I, II)

A solid-rocket test vehicle developed by General Dynamics. I was used especially to test the Mercury capsule, and II will be used to test the Apollo spacecraft.

LUNAR EXCURSION MODULE

The two-man vehicle that will land on the moon after the Apollo spacecraft enters lunar orbit.

LUNAR ORBITAL RENDEZVOUS

The concept for manned lunar landing adopted by NASA wherein the lunar excursion module leaves the spacecraft, lands on the moon, and later returns to the orbiting spacecraft. The excursion module will be jettisoned as the spacecraft leaves lunar orbit.

MANNED SPACECRAFT CENTER (MSC)

NASA headquarters responsible for development and operation of manned space vehicles (Mercury, Gemini, Apollo), located in Houston, Texas.

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MARSHALL SPACE FLIGHT CENTER (MSFC)	NASA operation responsible for design and development of space launch vehicles (Saturn, Advanced Saturn, Nova), located in Huntsville, Alabama.
MOCK-UP	A full-scale, three-dimensional representation of a complete spacecraft, individual module, and/or related equipment. Based on permanence and difficulty of alteration, mock-ups are graded as "soft," "semi-hard," and "hard."
MODULE	A combination of components, contained in one package or so arranged that together they are common to one mounting, which provides a complete function. (See command module, service module, lunar excursion module, etc.)
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION (NASA)	Civilian agency, sponsored by the U. S. Government, with research and development jurisdiction in aeronautical and space activities except those activities peculiar to or primarily associated with the development of weapon systems, military operations, or the defense of the United States.
NAUTICAL MILE (NM)	A measure of distance equal to 6, 076. 103 feet or approximately 1. 15 mile.
OBTAIN DATA	Denotes gathering engineering information that is to be measured to augment the general knowledge required in the development of the over-all spacecraft. This category may also be used for supplemental investigation, such as environmental studies, ground equipment studies, etc. The degree of instrumentation is not implied by this definition.
OPTICAL STAR TRACKER	A star tracker that locks onto the light of a particular celestial body. Distinguished from a radiometric star tracker. (See star tracker.)
OXIDIZER	A rocket propellant component, such as liquid oxygen, nitric acid, fluorine, and others, that supports the combustion of a fuel.

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PAD — A permanent or semipermanent load-bearing surface constructed or designed as a base upon which a launcher can be placed. Short for launch pad.

PITCH — The movement about an axis that is at once perpendicular to the Apollo longitudinal axis and horizontal to the Y-axis of the spacecraft.

PRE-QUALIFIED — A pre-qualified part or component is one that has been scheduled as the one most likely to succeed in the qualification program and will be used in production runs as well as during developmental test or early flight test prior to qualification.

PRESSURIZED SUIT — A garment designed to provide pressure upon the body so that respiratory and circulatory functions may continue normally, or nearly so, under low-pressure conditions, such as occur at high altitudes or in space without benefit of a pressurized cabin.

PROGRAMMED ROLL — An automatically controlled roll of a ballistic missile or satellite, usually executed during its vertical ascent before pitch-over.

PROGRAMMED TURN — The turn of a ballistic missile from vertical motion, after lift-off, to a curved path approximating the desired powered flight trajectory prior to the initiation of guidance.

PROPELLANT — A liquid or solid substance burned in a rocket for the purpose of developing thrust.

PROTOTYPE — A model suitable for complete evaluation of mechanical and electrical form, design, and performance. It is in final mechanical and electrical form, employing approved parts, and completely representative of the final equipment.

QUALIFICATION OF PART, COMPONENT, OR SYSTEM — A part, component, or system is considered qualified by definition after it has successfully completed all of the prescribed tests associated with relevant control specifications.

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READOUT — A radio transmitter transmitting data either instantaneously with computation of the data or by play of a magnetic tape on which the data have been recorded.

RECOVERY — The act of retrieving a portion of a launched missile or satellite that has survived reentry.

REENTRY — Return of a part of a space vehicle to the atmosphere after flight above the sensible atmosphere.

RELIABILITY — Reliability is the probability of performing without failure a specified function, under given conditions, for a specified period of time. It deals with the failure rates in time of specific items.

RETROROCKET — Relatively small rocket unit, usually solid propellant, installed on a rocket-propelled vehicle and fired in a direction opposite to the main motion to decelerate main unit.

ROLL — The movement of Apollo about its longitudinal (X) axis.

SATURN — The sun's sixth planet. A NASA rocket engine cluster in research and development expected to develop some 1, 500, 000 pounds of first-stage thrust. The Apollo launch vehicle.

SEPARATION — Moment when a full stage, half stage, a warhead, or a nose cone is separated from the remainder of the rocket vehicle; the moment when staging is accomplished.

SERVICE MODULE — Apollo module carrying propulsion equipment, fuel, reaction control systems, and communications power. It is used for thrust after booster separation, mid-course correction, lunar orbit, lunar orbit ejection, and earth return midcourse correction. It is jettisoned prior to reentry.

SERVICE PROPULSION SYSTEM — Engine and associated equipment providing thrust for service module functions. (See SERVICE MODULE.)

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SOFT LANDING
Landing on the moon or other spatial body at such slow speed as to avoid damage of landing vehicle. Soft landings on moon are anticipated by use of retrorockets.

SOLAR FLARE
Solar phenomenon that gives rise to intense ultraviolet and corpuscular emission from the associated region of the sun. This affects the structure of the ionosphere and interferes with communications.

SOLID PROPELLANT
A propellant in solid condition including all the ingredients necessary for sustained chemical combustion, such as a compound of fuel and oxidizer, usually in plastic caked form. It burns on its exposed surface, generating hot exhaust gases to produce a reaction force.

SPACECRAFT
In the Apollo program, any component or combined components of the flight vehicle not part of the launch vehicle: launch escape subsystem, command module, service module, adapter, or any combination of these.

SPECIFIC IMPULSE
The thrust produced by a jet-reaction engine per unit weight of propellant burned per unit time, or per mass of working fluid passing through the engine in unit time. It is equal to thrust in pounds divided by weight flow rate in pounds per second.

STABILIZATION AND CONTROL SUBSYSTEM (SCS)
An Apollo monitor system linked to navigation and guidance system, display system, and reaction control subsystem indicating spacecraft attitude (roll, yaw, or pitch).

STAGE
In a rocket vehicle powered by successive units, one or other of the separate propulsion units.

STAGNATION POINT
The location on a surface in an airstream where the air flow is zero.

STAR TRACKER	A telescopic instrument on a missile or spacecraft that locks onto a celestial body and gives guidance to the missile or other object during flight.
STATIC TESTING	Testing of a missile or other device in a stationary or holddown position, to verify structural integrity, to determine the effects of limit loads, or to measure thrust.
SYSTEMS ENGINEERING	Process of applying science and technology to the study and planning of an over-all aerospace vehicle system, whereby relationships of various parts of the system and the use of various subsystems are fully planned and integrated prior to time hardware designs are committed.
TELEMETERING	The technique of recording space data by radioing an instrument reading from a rocket to a recording machine on the ground.
THEODOLITE	A sighting and measuring telescopic instrument that gives a reading on horizontal or vertical angles.
TRAJECTORY	The path described by a missile or a space vehicle.
TRANSFER ELLIPSE	Path followed by a body moving from one elliptical orbit to another.
ULLAGE	The amount of fluid by which a tank falls short of being full; the loss through evaporation, spilling, or consumption; the amount remaining that cannot be drained from an emptied tank or container.
UMBILICAL	Any one of several electrical or fluid lines connected between the ground support operation and an uprighted rocket missile or space vehicle before launch.
UNSYMMETRICAL DIMETHYLHYDRAZINE (UDMH)	Rocket fuel which with aerozine will power the Apollo spacecraft.

VECTOR STEERING	Vernacular for a steering method where one or more thrust chambers are gimbal mounted so that the direction of the thrust force (thrust vector) may be tilted in relation to the center of gravity of the missile to produce turning.
VELOCITY VECTOR	Combination of two ballistic missile trajectory values: the speed of the missile's center of gravity at a designated point on the trajectory and angle between local vertical and the direction of the speed.
WHITE SANDS MISSILE RANGE (WSMR)	A proving ground in New Mexico under the control of the Army Ordnance Missile Command; supports Apollo abort tests.
YAW	Lateral movement of the Apollo spacecraft along the Z-axis in line of flight.

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